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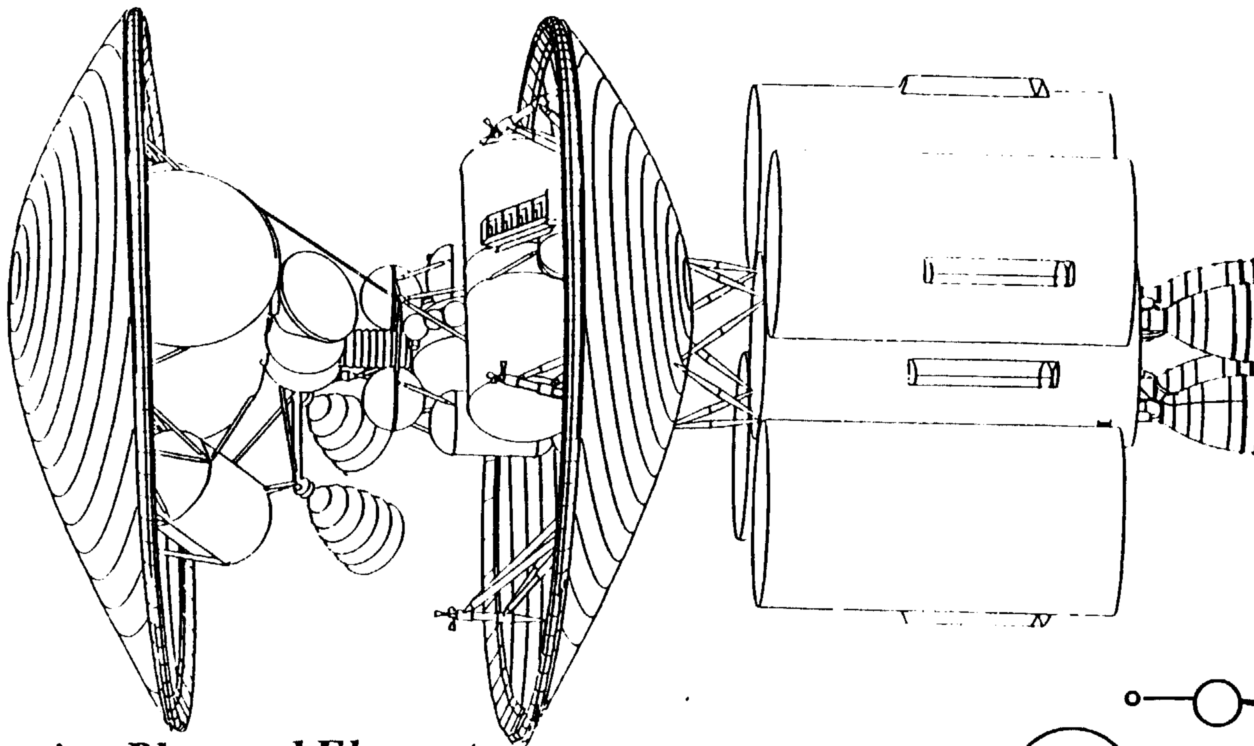

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Volume 2: Cryo/Aerobrake Vehicle

March 8, 1991

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ADVANCED CIVIL SPACE SYSTEMS

Boeing Aerospace and Electronics
Huntsville, Alabama

NASA Contract NAS8-37857

690

Space Transfer Concepts and Analyses for Exploration Missions

NASA Contract NAS8-37857

Cryogenic/Aerobrake Vehicle Implementation Plan and Element Description Document

Boeing Aerospace and Electronics
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Cryogenic/Aerobrake Implementation Plan and Element Description Document

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Huntsville, Alabama

Documentation Set:
D615-10026-1 IP and ED Volume 1: Major Trades, Books 1and 2
D615-10026-2 IP and ED Volume 2: Cryogenic/ Aerobrake Vehicle
D615-10026-3 IP and ED Volume 3: Nuclear Thermal Rocket Vehicle
D615-10026-4 IP and ED Volume 4: Solar Electric Propulsion Vehicle
D615-10026-5 IP and ED Volume 5: Nuclear Electric Propulsion Vehicle
D615-10026-6 IP and ED Volume 6: Lunar Systems

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Implementation Plan and Element Description Document
Cryogenic/Aerobrake
Table of Contents

Symbols, Abbreviations and Acronyms

ACRV	Advanced crew recovery vehicle
ACS	Attitude control system
AFE	Aerobrake Flight Experiment
A&I	Attachment and integration
Al	Aluminum
ALARA	As low as reasonably achievable
ALS	Advanced Launch System
ALSPE	Anomalously large solar proton event
am	Atomic mass (unit)
AR	Area ratio
ARGPER	Argument of perigee
ARS	Atmospheric revitalization system
art-g	Artificial gravity
asc	Ascent
ASE	Advanced space engine
AU	Astronomical Unit (=149.6 million km)
BIT	Built-in test
BITE	Built-in test equipment
BLAP	Boundary Layer Analysis Program
BFO	Blood-forming organs
C	Degrees Celsius
CAB	Cryogenic/aerobrake
CAD/CAM	Compter-aided design/computer-aided manufacturing
CAP	Cryogenic all-propulsive
C_d	Drag coefficient
CELSS	Closed Environmental Life Support System
CHC	Crew health care
CG	Center of gravity
C_L	Lift coefficient
cm	Centimeter = 0.01 meter
c/m	Crew module
CM	Center of mass
c/o	Check out
C of F	Cost of facilities
conj	Conjunction
COSPAR	Committee on Space Research of the International Council of Scientific Unions
CO2	Carbon dioxide
Cryo	Cryogenic
C3	Hyperbolic excess velocity squared (in km^2/s^2)
d	days
DDT&E	Design, development, testing, and evaluation
DE	Dose equivalent
deg	Degrees
desc	Descent
DMS	Data management system
dV	Velocity change (ΔV)

EA	Earth arrival
E arr	Earth arrival
Ec	Modulus of elasticity in compression
ECCV	Earth crew capture vehicle
ECWS	Element control work station
ECLSS	Environment control and life support system
EP	Electric propulsion
ESA	European Space Agency
e.s.o.	Engine start opportunity
ET	External Tank
ETO	Earth-to-orbit
EVA	Extra-vehicular activity
F_c	Circulation efficiency factor
FD&D	Fire Detection and Differentiation
F_{ew}	Life support weight factor
F_f	Specific floor count factor
F_{fa}	Specific floor area factor
F_i	Aerobrake integration factor
F_l	Specific length factor
F_n	Normalized spatial unit count factor
F_o	Path options factor
F_p	Useful perimeter factor
F_{pc}	Parts count factor
F_{pr}	Proximity convenience factor
F_{rp}	Plan aspect ratio factor
F_{rs}	Section aspect ratio factor
FSE	Flight support equipment
F_s	Vault factor
F_{ss}	Safe-haven split factor
F_u	Spatial unit number factor
F_v	Volume range factor
FY88	Fiscal Year 1988 (=October 1, 1987 to September 30, 1988. Similarly for other years)
g	Acceleration in Earth gravities (=acceleration/9.80665m/s^2)
GCNR	Gas core nuclear rocket
GCR	Galactic cosmic rays
GEO	Geosynchronous Earth Orbit
GN2	Gaseous nitrogen
GN&C	Guidance, navigation, and control
GPS	Global Positioning System
Gy	Gray (SI unit of absorbed radiation energy = 10^4 erg/gm)
hab	Habitation
HD	High Density
HEI	Human Exploration Initiative (obsolete for SEI)
HLLV	Heavy lift launch vehicle
hrs	Hours
hyg w	Hygeine water
HZE	High atomic number and energy particle
H2	Hydrogen
H_2O	Water

ICRP	International Commission on Radiation Protection
IMLEO	Initial mass in low Earth orbit
in.	Inches
inb	Inbound
IP&ED	Implementation Plan and Element Description
IR&D	Independant research and development
Isp	Specific impulse (=thrust/mass flow rate)
ISRU	In-situ resource utilization
JEM	Japan Experiment Module (of SSF)
JSC	Johnson Space Center
k	klb
keV	Thousand electron volt
kg	Kilograms
klb	Kilopounds (thousands of pounds. Conversion to SI units=4448 N/klb)
klbf	Kilopound force
km	Kilometers
KM	Kilometers
KM/Sec	Kilometers per second
KM/SEC	Kilometers per second
ksi	Kilopounds per square inch
L/D	Lift-to-drag ratio
LD	Low density
LDM	Long duration mission
LEO	Low Earth orbit
LET	Linear energy transfer
LEV	Lunar excursion vehicle
LEVCM	Lunar excursion vehicle crew module
Level II	Space Exploration Initiative project office, Johnson Space Center
LH2	Liquid hydrogen
LiOH	Lithium hydroxide
LLO	Low Lunar orbit
LM	Lunar Module
LOR	Lunar orbit rendezvous
LOX	Liquid oxygen
LS	Lunar surface
LTV	Lunar transfer vehicle
LTVCM	Lunar transfer vehicle crew module
L2	Lagrange point 2. A point behind the Moon as seen from the Earth which has the same orbital period as the moon.
m	Meters
[MarsGram	Western Union interplanetary telegram]
[MARSIN	Martian pornography]
MASE	Mission analysis and systems engineering (same as Level II q.v.)
MAV	Mars ascent vehicle
M/C_DA	Ballistic coefficient (mass / drag coefficient times area)
MCRV	Modified crew recovery vehicle
m_e	Mass of electron
MEOP	Maximum expected operating pressure
MeV	Million electron volt

MEV	Mars excursion vehicle
MLI	Multi-layer insulation
mm	Millimeter (=0.001 meter)
MMH	Monomethylhydrazine
MMV	Manned Mars vehicle
MOC	Mars orbit capture
MOI	Mars orbit insertion
mod	Module
M&P	Materials and processes
MPS	Main propulsion system
MR	Mixture ratio
m/sec	Meters per second
MSFC	Marshall Space Flight Center
Msi	Million pounds per square inch
mt	Metric tons (thousands of kilograms)
mT	Metric tons
MTBF	Mean time between failures
MTV	Mars transfer vehicle
MWe	Megawatts electric
m^3	Cubic Meters
N	Newton. Kilogram-meters per second squared
n/a	Not applicable
NASA	National Aeronautics and Space Administration
NCRP	National Council on Radiation Protection
NEP	Nuclear-electric propulsion
NERVA	Nuclear engine for rocket vehicle application
NSO	Nuclear safe orbit
NTR	Nuclear thermal rocket
N2O4	Nitrogen tetroxide
OSE	Orbital support equipment
OTIS	Optimal Trajectories by Implicit Simulation program
outb	Outbound
O2	Oxygen
PBR	Particle bed reactor
Pc	Chamber pressure
PEEK	Polyether-ether ketone
PEGA	Powered Earth gravity assist
P/L	Payload
POTV	Personnel orbital transfer vehicle
pot w	Potable water
PPU	Power processing unit
prop	Propellant
psi	Pounds per square inch
PV	Photovoltaic
Q	Heat flux (Joules per square centimeter)
Q	Radiation quality factor
RAAN	Right ascension of ascending node
RCS	Reaction control system

Re	Reynolds number
RF	Radio frequency
RMLEO	Resupply mass in low Earth orbit
RPM	Revolutions per minute
RWA	Relative wind angle
R&D	Research and Development
	Rendezvous and dock
SAA	South Atlantic Anomaly
SAIC	Science Applications International Corporation
SEI	Space Exploration Initiative
SEP	Solar-electric propulsion
SI	International system of units (metric system)
SiC	Silicon carbide
SMA	Semimajor axis
sol	Solar day (24.6 hours for Mars)
SPE	Soalr proton events
SRB	Solid Rocket Booster
SSF	Space Station Freedom
SSME	Space Shuttle Main Engine
STCAEM	Space Transfer Concepts and Analysis for Exploration Missions
stg	Stage
surf	Surface
Sv	Sievient (SI unit of dose equivalent = Gy x Q)
S1	Distance along aerobrake surface forward of the stagnation point
S2	Distance along aerobrake surface aft of the stagnation point
S3	Distance along aerobrake surface starboard of the stagnation point
t.	Metric tons (1000kg)
TBD	To be determined
Tc	Chamber temperature
TCS	Thermal control system
TEI	Trans-Earth injection
TEIS	Trans-Earth injection stage
t.f.	Tank weight factor
THC	Temperature and humidity control
TMI	Trans-Mars injection
TMIS	Trans-Mars injection stage
TPS	Thermal protection system
TT&C	Tracking, telemetry, and control
T/W	Thrust to weight ratio
UN-W/25Re	Uranium nitride - Tungsten/25% Rhenium reactor fuel
VAB	Vehicle Assembly Building
VCS	Vapor coolled shield
Vinf	Velocity at infinity
WBe_2C/B_4C	Tungsten beryllium cabide/Boron cabide composite
WMS	Waste management system
W/O	Without
WP-01	Work package 1 (of SSF)
w/sq cm	Watts per square centimeter (should be Wcm^{-2})

Z	Atomic number
zero g	An unaccelerated frame of reference, free-fall

[order: numbers followed by greek letters]

100K	≤100,000 particles per cubic meter larger than 0.5 micron in diameter
7*n*7	Where n=(0,2-6): Boeing Company jet transport model numbers
°k	Kelvin (K)
+e	Positive charge equal to charge on electron
-e	Charge on electron
ΔV	Change in velocity
s	Standard deviation
μg	Microgravity

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I. Evolution of Concept

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Concept Development

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EVOLUTION OF THE CRYOGENIC PROPULSION VEHICLES

TECHNICAL ARCHITECTURE PRESUMED LEVEL I REQUIREMENTS - During the course of the STCAEM study, and particularly during the *90 Day Study*, many SEI (then HEI) transportation requirements were generated by Office of Exploration Level II. These are reported as appropriate and necessary in various sections of this report, as well as in the *STCAEM Implementation Plan & Element Description Document* technical volumes. Here, space only permits a summary discussion of the Level I requirements adopted by STCAEM as they evolved during the course of the study. The concepts developed and analyzed ultimately were to accommodate the in-space transportation functions required to support the buildup of a permanent presence on the Moon and initial human exploration of Mars. Thus, our Level I requirement was simply to **deliver cargo reliably to the surfaces of the Moon and Mars, and to get people to those places and back safely.** Vehicles in support of missions to *other* destinations are not part of SEI *per se*, and were not addressed by STCAEM. Planet surface system characteristics and Earth-to-orbit (ETO) launch vehicle characteristics were adopted as needed for manifesting purposes, largely intact from other sources. No design work was performed for these two categories. In addition, the mission planning horizon was limited to the year 2025, about 35 years from now.

The chief Level II requirement governing the dimensions of the vehicle concepts we developed came to us during the *90 Day Study*, and was a crew size of 4 for Mars missions. Subsequently, STCAEM performed a simple skill mix analysis or these long-duration missions. Our result was that doubling up on critical skills (for redundancy), given reasonable expectations of how many skills each crew member could become expert in, requires in fact a minimum of 6 - 7 crew members for Mars missions. For the sake of consistency, our vehicle concepts are shown comparable to the *90 Day Study* results, sized for four crew. Impacts accruing from larger crew sizes are discussed in the Major Trades IP&ED book.

CONCEPT DEVELOPMENT METHODOLOGY - A vehicle concept emerges gradually through the iterative combination of requirements analysis, subsystems analysis, mass synthesis, performance analysis and configuration design. Because of the cascading, cause-and-effect nature of specific technical decisions in this cyclic process, the ability for a particular concept to remain fully parametric is incrementally lost, sacrificed for depth of detailing. The need to penetrate deeply even at the conceptual stage is twofold: (1) to uncover subtle integration interactions whose ramifications fundamentally revise the concept as they reflect back up the information hierarchy; and (2) to enable the production of graphical images of the concepts capable of being communicated widely *but grounded firmly in engineering detail.* If circumstances allow the concept development process to engage many cycles of reflexive adjustment, from requirements all the way down through subsystem detailing, the design oscillations subside eventually and the product that emerges is a robust and defensible concept. Basic differences in problems posed and solutions engineered lead concept developments in different directions. "Like" problems and solutions gravitate together; their recombination and resolution results in distinct, identifiable vehicle concepts which constitute *vehicle archetypes.* A concept is archetypal if it spawns concept progeny whose ancestry is clear, and if in so doing its salient features recognizably survive subsequent refinement, development and scaling. The ultimate purpose of the STCAEM Concepts and Evolution tasks was to generate, analyze, evaluate and describe such vehicle archetypes, and the role they could play in human space exploration missions.

The STCAEM architecture analysis identified seven major classes of transportation architecture for SEI lunar and Mars missions. Some are derived from different propulsion technology candidates; some are derived from distinct mission philosophies independent of propulsion method; most have many sub-options. Vehicle archetypes are keyed more closely to propulsion method than to mission mode, however, so we found that all seven SEI transportation architectures can be accomplished by derivative combinations of just five archetypal Mars transfer vehicle (MTV) concepts, two archetypal Mars excursion vehicle (MEV) concepts, and one archetypal lunar transportation family (LTF) concept. The concept evolution of these archetypes is outlined in Section x.2.

DESIGN AND NECKDOWN CRITERIA - STCAEM concept development was punctuated by four "neckdowns", which winnowed down the option candidates generated at each successive level of detail throughout the study. The four neckdowns were intended to result in: (1) **feasible options**, based on promising propulsion technologies capable of performing SEI-class missions; (2) **preferred options**, representing the handful of candidates whose performance and technological readiness were judged to warrant detailed study; (3) **integrated concepts**, vehicle archetypes developed sufficiently to uncover their major integration concerns and architectural context ; and (4) **detailed concepts**, based on the reconciled integration of traded subsystems. The *90 Day Study* occurred such that the first two neckdowns were effectively reversed; cryogenically propelled, aerobraking technology was necessarily preferred at that time, due to depth of understanding. However, STCAEM later rounded out the picture by completing all four neckdown activities, in an ongoing manner throughout the study.

Studying the program architecture implications of various technology options for SEI missions led to the conclusion that the most generally accessible discriminators, *cost* and *risk*, are driven by more subtle technical discriminators than, for instance, initial mass in low Earth orbit (IMLEO). These can be grouped into three broad categories: *feasibility*, *flexibility*, and *multi-use design*. As indicated above, feasibility was the first filter for all concepts considered by STCAEM. **Flexibility** has three components: (1) *robustness*, which is the ability to perform nominally despite variable or unanticipated conditions; (2) *resiliency*, which is the ability to recover from accidental delays or mishaps; and (3) *evolution*, which is an adaptation over time to changing requirements. Flexibility is thus a measure of a program's technical strength and safety in the face of variable extrinsic factors. **Multi-use design** has two components: (1) *re-usability*, which means using the same hardware item more than once; and (2) *commonality*, which means using the same hardware design in more than one setting. Multi-use design is thus a measure of a program's cost-effectiveness and intrinsic longevity. These two key architecture drivers were paramount in interpreting the results of STCAEM's technical trade studies, and figured prominently in the development of element concepts.

MARS TRANSPORTATION - Four Mars transfer propulsion candidates survived all STCAEM neckdowns: cryogenic chemical, nuclear thermal, nuclear electric, and solar electric. Analysis of aerobraking resulted in two performance ranges of interest for Mars entry (hypersonic L/D = 0.5, and L/D = 1.0), as well as the use of high-energy aerobraking (HEAB) for capture at Mars. Consequently, the five archetypal MTV concepts are based respectively on: cryogenic/aerobraking (CAB), cryogenic all-propulsive (CAP), nuclear thermal rocket (NTR), nuclear electric (NEP), and solar electric (SEP) propulsion technologies. The two archetypal MEV concepts are based on the "low" and "high" L/D performance ranges analyzed.

<u>Cryogenic/Aerobraked Mars Transfer Vehicle (CAB)</u> - NASA selected cryogenic chemical propulsion, augmented by aerobraking for capture and landing at Mars, as the opposition-profile baseline for the *90 Day Study*. The archetype which first resolved the dominant configuration complications for CAB Mars missions already existed (Boeing, 89). With this foundation the *90 Day Study* was able to progress rapidly into performance, subsystem, operations and programmatics analyses. The *90 Day Study* exercise in turn enabled refinement and validation of the CAB archetype. The major drivers for the CAB archetype are:

1) High-thrust chemical propulsion: engine-out design to accommodate shifting vehicle mass center as the mission progresses, given the fact of engine clustering and limited gimbal angle; propulsion system geometry for in-flight testing before critical mission maneuvers; and avoidance when possible of aerobrake penetrations.

2) High-energy aerobraking: current understanding of aftbody wake closure geometry, and aerodynamic simulation-based constraints on mass center location; mutual independence of MEV and MTV during final approach to Mars space, since each is captured separately; packaging of the entire MTV system in as small a capture aerobrake as possible; potential requirement for MTV brake retention and re-use for Earth capture upon return.

3) Rotating artificial gravity: physiological constrains drive the CAB archetype toward deployable tether schemes because of the effort to make the aerocaptured vehicles as compact as possible. This makes the physical arrangement of the MTV systems difficult, given both a requirement to maintain all habitable volumes (both the MTV habitat and the MEV crew cab and surface module) contiguous during transfer, and the fact that the only rotation countermass available on the return leg is the empty MTV TEI propulsion system.

4) Modular vehicle design, in an effort to maximize system commonality, to standardize integration and operations protocols, and especially to accommodate the widely varying energy (propulsive) requirements of opposition-class missions. In STCAEM, opposition missions were designed to collect most of the energy difference in the TMI ΔV. This burden was more easily accommodated by the TMIS, which became a highly modular vehicle system.

5) Robotic-mediated operations: facilitating machine access into the densely packaged systems of the CAB vehicle , and designing provision for robotic EVA maintenance during the mission, is a tough but essential requirement. We baselined an operations concept in which manipulator systems could travel around the rims of the rigid aerobrake structures, both to assist in assembling the vehicles at Earth and to service them *en route*.

A concept called the "Shuttle-Z 3rd Stage" was detailed in response to a Level II trade. This is a modular version of the TMIS, in which each section uses its engine twice (once for ETO orbit insertion and again for the Mars departure burn). The fundamental problem with the scheme is that, with engines located on each TMIS section instead of clustered in the center, mass-balanced engine-out on TMI is not possible without the addition of an extremely long (120 m) truss to separate the TMIS from the payload mission vehicles.

A configuration trade analysis revealed that avoiding the need for Mars orbit rendezvous upon arrival between a separate MEV and MTV by configuring one large, aerocaptured vehicle was not practicable (either a *very* large aerobrake, or a reconfigurable cryogenic propulsion system, appeared necessary).

An Earth-Mars cycler vehicle capable of providing periodic transfers between the two planets is one potential mission mode addressed by our architecture assessment. Such a vehicle could take a variety of forms, but for SEI-class missions, the basic function could be accomplished with a variation on the CAB vehicle. For the conventional cycler profile, aerocapture energies for the "taxi" craft needing to get into parking orbits at Mars are quite high. Re-usable vehicles for this job would probably require heavy and/or complex thermal protection systems.

OTHER SYSTEMS:

Cryogenic All-Propulsive Mars Transfer Vehicle (CAP) - The CAP archetype is fundamentally a variation of the CAB archetype, but is reported here as a separate archetype because its mission philosophy is quite distinct. The CAP concept was developed in response to two drivers:

1) Exploration of alternative purposes for SEI Mars missions led, after the *90 Day Study*, to more in-depth discussions of the merits of conjunction *vs.* opposition profiles. Initial presumptions favored short total mission durations; this approach remained typical after the FY88 and FY89 OEXP study cycles, in which very short, compressed opposition or "split-sprint" mission modes figured prominently. However, given the 30 - 60 d Mars staytime realistically permitted by their astrodynamics, the ratio of usable surface time to total mission time for opposition profiles is about 10 %. After the *90 Day Study*, this was recognized more widely as a relatively disappointing science return on a large engineering investment, exacerbated by the possibility of extrinsic events (like Martian dust storms) precluding landing altogether. By comparison, the same ratio for a typical conjunction mission is about 30 %. The top-level costs associated with exploiting the greater opportunity to do in-depth science proffered by conjunction missions are two: (1) the requirement for more elaborate surface payload manifests to support both that science and the crews to conduct it for year-long stays; and (2) the greater risk to mission completion incurred by having the crews and hardware spending almost 3 yr in deep space instead of about 1.5 yr.

The conjunction profile offers other benefits recognized much later. First, the opportunity variation in mission energy requirements is much reduced for the conjunction case, so that mission hardware can be more consistent from one opportunity to the next. This would minimize the actual program upset resulting from a missed opportunity. Second, having of order 300 d available at Mars would permit more flexible mission design. For example, rather than spending the entire staytime on the surface, the mission might carry multiple landers each destined for short visits to widely separated surface sites (or crew rescue at a given site). And finally, although conjunction missions are roughly twice the length of opposition missions, the bulk of that difference *can* consist of time spent on the surface of Mars, under the radiation shielding afforded by the martian atmosphere. The actual in-space transfers are about equal in length outbound and inbound, and their total is less than the total in-space transit time for typical opposition missions. Thus in scenarios required to minimize astronaut exposure to in-space galactic cosmic radiation (GCR), well-designed conjunction missions are of great interest. (Trip times can be shortened further still, until the so-called "conjunction fast transfer" mission energy requirements approach those for opposition missions.)

Conjunction low-energy missions do not benefit from HEAB, so these missions need only carry aerobrakes for entry and landing. Performing Mars capture with cryogenic chemical propulsion leads to three fundamental distinctions between CAP and CAB concepts:

1) The MTV and MEV(s) are captured together, precluding the possibility of failure to rendezvous and consequent scrub of landing attempts.

2) The Earth-departure (TMI) stage grows into a multi-staged propulsion stack, with TMIS, deep-space burn (DSB) stage, and Mars arrival (MOC) stage. This changes the overall aspect ratio of the all-up vehicle, making it longer, which has implications for attitude control and debris shielding in LEO.

3) Relaxing the requirement for the MTV to be an aerobraked vehicle means that the systems constrained in the CAB case to be packaged behind an aerobrake can be distributed differently. Thus the Mars-departure (TEI) propulsion system can be combined with the MOC system and placed at the opposite end of the vehicle from the MTV habitation system and payload. This in turn means that rotating artificial gravity can be accomplished as simply as for the NTR vehicle, by configuring a long, lightweight truss between the propulsion end and the payload end, and spinning this rigid assembly end-over-end. Tethered solutions are not required because aerobrake

packaging is no longer a problem. This last set of CAP consequences departs from the CAB concept sufficiently for their resolution to constitute a distinct vehicle archetype.

ARTIFICIAL GRAVITY (CAB) - The need for artificial gravity on long-duration interplanetary transfers has not been established. Neither has the *lack* of such a need, however, so STCAEM was obligated to examine the penalties incurred by requiring continuous artificial gravity *en route* between Earth and Mars. Various approaches to rotating artificial gravity have been proposed; STCAEM assessed all of them, and invented some new ones. The fundamental *design* problems associated with artificial gravity derive from: (1) the need for a countermass for rotation; and (2) the high mass cost of precessing the angular momentum vector of a system having large rotational energy. Elegant solutions to both are elusive, and vary widely with propulsion option. Secondary complications are communications and navigation pointing, flight structures sized to hang heavy vehicles, and possibly material fatigue. The fundamental *operations* problems associated with artificial gravity involve crew EVAs during rotation, robotic maintenance in the vehicle's gravity field, crew physiological and psychological responses to a rotating environment, performing minor course-correction propulsive maneuvers and testing the capability prior to departure. Our work has verified that artificial gravity appears feasible for Mars-class missions, for all propulsion options, at fairly modest mass penalties.

The CAB archetype involves more complexity. The MTV habitat must be contiguous with the MEV crew modules, and yet for the return trip the (empty) MTV propulsion system is the only available countermass to the MTV habitat. Thus the MTV hab and the MTV propulsion system must be separated by a few hundred meters; however, the entire MTV must also package behind an aerobrakefor capture at Mars. One solution we rejected for mass and habitability reasons splits the transfer habitat system in two halves, held when not aerobraking at opposite ends of a deployable tunnel. A more sensible approach is to use tethers, configuring the MTV systems such that they are properly mass-balanced for propulsive burns and aerocapture, but can slip apart as the tethers are unreeled for artificial gravity. The center of rotation provides a convenient location for a despun power/navigation/communications utility.

ARTIFICIAL GRAVITY (CAP) - .The CAP and NTR archetypes accommodate artificial gravity easily. Both are high-thrust systems, so their burn times are extremely short (minutes to hours) compared to coasting transfer time (months). Critical propulsion maneuvers can occur during nonrotating periods of microgravity, at the cost only of spinup/spindown propellant. In general, the propulsion system remaining through the end of the mission can serve as countermass to the contiguously connected habitation systems. When separated by a lightweight truss, they can just spin end-over-end during coast phases to provide sufficient gravity at a comfortable spin rate with acceptable vestibular disturbance (we baselined 1 g to insure full conditioning for surface activity upon arrival at Mars, and 4 rpm maximum spin rate, which together lead to a 56 m separation between the hab and the center of mass). The additional mass of the truss and propellant for a few budgeted spinup/spindown cycles is of order 10 % of IMLEO.

Low-L/D Mars Excursion Vehicle (MEV) - The MEV archetype development began during, and was resolved just following, the NASA *90 Day Study*. It was originally conceived as a means of delivering 25 t of undefined payload to the surface of Mars. However, the specification of crew cab provisions, the analysis of vehicle mass balance, and consequently the configuration design of the vehicle all depend on specifics of the payload manifest. We assumed a 20 t reference surface module as an integral part of the MEV. This led to a "Mars campsite" design intended to support a crew of four for 30 - 60 d and became or standard lander design. Chief departures from the lunar campsite mode of operation were:

1) The MEV arrives with the crew already onboard, and so is capable of a really self-contained mission.

2) The MEV also brings with it an ascent vehicle (MAV) with a separate propulsion system, configured optimally for the ascent phase (or ascent after breakaway from the descent stage during a descent abort). The crew cab for the MAV is the operations bridge for the MEV during all its mission phases.

3) The MEV is configured for packaging within an L/D = 0.5 aerobrake. For CAB missions, this brake captures the as-yet unmanned MEV into Mars orbit autonomously, before rendezvous with the MTV, and is used again for the descent. For CAP and other types of missions with propulsive Mars orbit capture, this brake is used only for descent. In all design cases, terminal descent engines are extended through ports in the windward surface of the brake at low Mach number, and the brake is jettisoned subsequently, prior to touchdown.

The MEV configuration was developed to permit later removal and relocation of the surface habitat module, with the aid of surface construction equipment. A variant of the MEV, without either surface module or MAV, was analyzed for delivery of heavy cargo on unmanned missions. A quick assessment was made of the feasibility of re-using an MEV, presuming *in situ* production of oxygen and retention of the aerobrake until touchdown. The outcome was positive, although: (1) additional brake hatches appeared necessary for landing gear deployment, crew egress, and cargo offloading; and (2) a lightweight top-shroud appeared advisable due to aerodynamic drag on ascent, and to permit the crew bridge to protrude beyond the presumed wake-protection limit for direct surface viewing during terminal approach. Configuration options for a "split-stage" MEV, in which the same, or a portion of the same, propulsion system is used for ascent as for terminal descent, were also investigated, and shown to be simple variations of the archetype.

Our baseline aerobrake assembly concept presumed robotic-mediated final assembly of pre-finished, rigid aerobrake segments at *Freedom*. Packaging such segments efficiently by nesting them in an ETO launch shroud is made challenging because of: (1) the aerobrake's asymmetrical, deep-bowl shape, in which the maximum depth of a typical "slice" is comparable to reasonable shroud diameters; and (2) the aerobrake's lip, required for both aerodynamic performance and structural stiffening around the free brake edge. Subsequent manifesting analysis, in which segments were configured according to an initial rib-and-spar structure concept, indicated that two ETO flights would be required to launch a single aerobrake in several pieces. Such extremely volume-limited and volume-inefficient manifesting is an unacceptably poor use of the expensively developed capability that a heavy-lift ETO system represents.

In response to this manifesting problem, STCAEM proposed the "integral launch" concept, in which a fully assembled, integrated aerobrake is launched externally, mounted on the side of the launch vehicle exactly analogous to current STS operations. The low-L/D brake is comparable to the STS orbiter in linear dimensions, and is light enough to launch two at once, with capacity to spare for other, shrouded payload as well. Ascent performance of such a flight configuration requires study; the critical question is whether ascent loads would size the aerobrake structure out of the competitive mass range for the mission itself.

Our structural analysis indicates that since the deep bowl-shaped aerobrake loads like a doubly-curved shell, it may be possible to construct an actual "aeroshell" without resorting to ribs and spars or some other articulated skeletal structure system. The shell would be made of a relatively thin honeycomb-type material system with integral TPS. However, lip buckling would still require a stiff rim, probably facilitated by a closed-tube-section structure. Such a brake may be lighter, and certainly simpler, but the thickened rim would still cause packaging problems due to nesting interference.

High-L/D Reusable Mars Excursion Vehicle (RMEV) - The RMEV archetype development occurred in response to three drivers:

(1) Analysis so far indicates that L/D = 0.5 is sufficient at Mars for controlling an aero-vehicle at Mars. However, the existence of some mission design studies in the literature which advocate L/D > 1.5 for Mars, combined with our preliminary understanding of controllability under Mars conditions, make it important to know in detail how different the configuration constraints imposed by higher L/D would be from those imposed by the lower L/D (which by 1989 had come to be regarded generally as appropriate).

2) As the *90 Day Study* stimulated thinking about what the purpose of SEI Mars surface missions should be, concern developed that global, or at least wide, access to the surface of Mars was potentially important. High-thrust Mars transfer propulsion systems (chemical or NTR) tend to be mass-constrained by arrival and departure vector geometry to certain parking orbit conditions. Although there is no lack of interesting (scientifically important) landing sites accessible from the periapsis of *any* orbit at Mars, the fact that performance-optimized parking orbits are unique for each high-thrust opportunity causes a site-access problem if returning to the same surface site is required (for base buildup). Thus for high-thrust transfer propulsion options particularly, an ability to achieve cross-range on lander entry may be important. High L/D enables greater cross-range capability.

3) Certain Mars lander issues not imposed as requirements during the *90 Day Study* required analysis and design validation. Developing a new MEV concept, substantially different from the baseline MEV, allowed us to investigate those issues simultaneously and thoroughly. Specifically, we addressed: (1) a deep aerobrake structure concept, of interest for maximum structural efficiency and therefore reduced brake mass; (2) the ability to deliver large-envelope cargo manifests, represented in our design by a long-duration surface habitat module sized for 10 crew; and (3) re-usability of the MEV, based on *in situ* production of cryogenic propellant.

The vehicle shape represented by the RMEV has applications for other interesting mission modes, concepts for which have yet to be investigated in detail. Three examples are: (1) a smaller RMEV, sized commensurately with the MEV to be a modest cargo-delivery vehicle; (2) a direct-landing MTV, whose return propellant would be manufactured *in situ* on Mars; and (3) re-usable aerobraked "taxi" vehicles capable of performing the Earth-Mars cycler embark/debark function.

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Cryogenic/Aerobrake (CAB)
Reference Configuration

Introduction

The cryogenic/aerobrake (CAB) concept was used as the NASA 90-day Study reference vehicle. It offers conceptual continuity with the mainstream Mars transportation studies performed over the last several years. Its only major new technology development is high energy aerobraking (HEAB) for planetary capture, but the concept also requires a high-thrust cryogenic space engine. Being able to land on Mars using the CAB concept requires a successful rendezvous between separately captured vehicles in Mars orbit.

Nominal Mission Outline

- The vehicle is assembled, checked out and boarded in LEO
- The TMI burn occurs and the TMIS is jettisoned
- MTV/MEV coasts to Mars
- MTV and MEV separate 50 days prior to Mars capture
- The MEV aerocaptures robotically a day ahead of the MTV, providing last-minute verification of atmospheric conditions and targeting
- The MTV captures, followed by rendezvous in the parking orbit with the MEV
- The landing crew transfers to the MEV and checks it out
- The MEV descends to the surface, jettisoning its aerobrake prior to landing
- After surface operations, the ascent vehicle (MAV) leaves its descent stage and surface payloads, ascends to orbit and docks with the MTV for crew transfer
- The MAV is jettisoned in Mars orbit, and the TEI burn occurs
- The MTV coasts back to Earth
- The crew transfers to a modified ACRV (MCRV), jettisons the MTV and performs a direct entry at Earth (optional: the entire MTV aerocaptures into a LEO parking orbit for refurbishment and re-use)

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Vehicle Systems

The vehicle consists of three main elements: the Mars Excursion vehicle (MEV), the Mars Transfer Vehicle (MTV) and the Trans-Mars Injection Stage (TMIS).

Mars Transfer Vehicle (MTV)

The MTV configuration shown consists of a transit habitat sized for 4 crew, an aerobrake, and a TEI propulsion system. The transit hab is located centrally in the aerobrake with an external airlock and an MCRV attached to the top (in the configurations shown, an Apollo-style ECCV was used to represent the MCRV). The airlock allows access to the MEV crew cab and surface habitat during all phases of the transfer mission until the MEV separation 50 days prior to Mars arrival. The MCRV is used for mission scenarios featuring direct-entry crew return; these scenarios expend the entire MTV upon return to Earth. In a reusable mode, the entire MTV would be aerocaptured back at Earth for refurbishment and re-use; a second airlock would be located in place of the MCRV. The aerobrake is of identical geometry and construction as the MEV aerobrake, but is stronger and heavier due to its larger payload mass, and does not require any engine doors. The propulsion system (TEI) is divided symmetrically into two tank-stacks straddling the transit hab, like the MAV tankset configuration. The propulsion system is oriented at an angle relative to the aerobrake axis, with the two engines aimed out the rear of the aerobrake, to avoid TPS penetrations while still permitting mass-balanced operation during the burn.

Trans-Mars Injection Stage (TMIS)

The TMIS consists of a core unit with four advanced space engines (ASE), avionics and cryogenic propellant tanks, and provision for up to four "strap-on" propellant tanksets. This configuration allows propellant cross-feeding in the case of engine-out, and modular accommodation of the entire stage's performance according to the mission opportunity requirements. Keeping the engines close together on the core stage allows tracking the CM during an engine-out condition via gimballing. This strategy avoids either opposite-shutoff (leading to long burn times and greater gravity losses), or a requirement for extra structure (a 125m truss) between the propellant tanks and engines to allow CM tracking. The TMIS accounts for about 75 % of the total IMLEO, a substantial per-mission resupply cost.

Mars Excursion Vehicle (MEV)

The reference MEV is a manned lander that can transport a crew of 4 to the surface. It consists of a surface-stay habitat module (roughly SSF-module size), an airlock, 5 t of surface-science payload, a cryogenic descent propulsion system with four engines and bus structure, and the ascent vehicle (MAV). The MAV consists of a short-duration crew cab, and cryogenic ascent propulsion system with two engines. All propellant tanks are mass-balanced around their maneuver CMs so that no lateral CM shifting occurs. The entire MEV is packaged in a rigid, truncated-hyperboloidal aerobrake with L/D = 0.5, to which it is attached at eight points (four bus-frame corners and four landing-gear footpads). The aerobrake is fitted with doors which open to allow the descent engines to extend and ignite prior to aerobrake separation (allowing full benefit of the brake's drag). The brake is then jettisoned as the landing gear extend prior to terminal approach and hovering touchdown.

Dominant configuration constraints for the MEV are as follows:

•Payload manifesting
•Surface access
•Crew visibility
•Contigurous crew volumes
•Short vehicle stack
•Engine-out capabilities
•On-orbit assembly

Payload manifesting is mainly a proximity and mass balance issue. The surface habitat and airlock, which is the bulk (80%) of the payload, require access to the ascent crew cab and the surface, as well as being mass balanced for proper flight. The science payload requires surface access for ease of unloading. Docking is

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facilitated by placing the crew cab high in the vehicle stack. The flight deck window is located to provide viewing to the surface for landing as well as to the upper hatch for docking. Keeping crew volumes contiguous allows access during flight for check-out procedures and simulation training. The vehicle stack is kept as short as possible for aerobrake wake protection, which tends to conflict with having the center of mass (CM) as high as possible, desirable for a small engine gimbal-angle to provide minimal steering loss in an engine-out scenario. A high CM within a short stack is accomplished by placing the dense ascent LOX high in the configuration. Finally, although the dominant constraints for the MEV derive from its performance at Mars, consideration has been given to its ETO launch. It is configured to be launched in a few, large, pre-integrated systems for minimal on-orbit assembly. For example, the ascent vehicle can be launched intact in a 10 m diameter shroud, while the descent structure can be launched in 2 sections for fairly simple on-orbit assembly and integration.

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Mars Mission Vehicle in LEO

The Mars vehicle LEO configuration is shown here ready for trans-Mars insertion (TMI).

The TMI stage launches the vehicle out of Earth orbit on a trans-Mars trajectory. There are four propellant tanks and five engines in the TMI stage; it is modularized for compatibility with the launch vehicle. The elements of the TMI stage are launched fully loaded with propellant.

The Mars excursion vehicle includes an aerobrake for Mars capture and entry/landing, a descent propulsion stage, an ascent propulsion stage with crew module for Mars descent, ascent, and contingency surface operations, and 25 t. of surface payload (a habitat and science) for normal surface operations.

The Mars transfer vehicle includes its own aerobrake for Mars capture, a long-duration crew habitat for the trips to and from Mars, a propulsion system for boost out of Mars orbit to return to Earth, and the Earth crew capture vehicle. The TMI stage is bookkept as part of the Mars transfer vehicle for WBS purposes. On some missions, the MTV aerobrake returns to Earth with the vehicle so that the MTV (except for the TMIS) can be captured in Earth orbit for reuse on another mission.

All crew volumes are contiguous between the MEV and MTV during TMI and coast.

The mass totals for option 1 and 5 are shown for comparison. The only difference between options 1 and 5 is that option 5 carries a surface reconnaisance vehicle into Mars orbit on the MEV (it is not shown on the chart). The surface reconnaisance vehicle is launched from the Mars parking orbit to perform robotic exploration of a future human landing site.

Cryo/Aerobraking

Trades and Rationale

- A core stage with four advanced engines and four "plug-in" propellant tanks. Tanks and core stage rendevous and dock automatically. Core stage provides simple plumbing and good engine out performance.

Mission Modes And Operations

- NASA 90 day study baseline.
- Vehicle assembled in SSF orbit.
- TMIS abandoned after TMI burn.
- MEV/MTV separate prior to Mars aerocapture.
- Crew transfer to MEV/Aerobrake after MTV/MEV rendevous.
- MEV/Aerobrake entry. Aerobrake jettisoned prior to landing.
- Crew cab ascent after surface mission, leaving lander and surface hab.
- Crew cab left in Mars orbit after rendevous, docking and crew transfer.
- TEI burn.
- Crew return to SSF after aerocapture.

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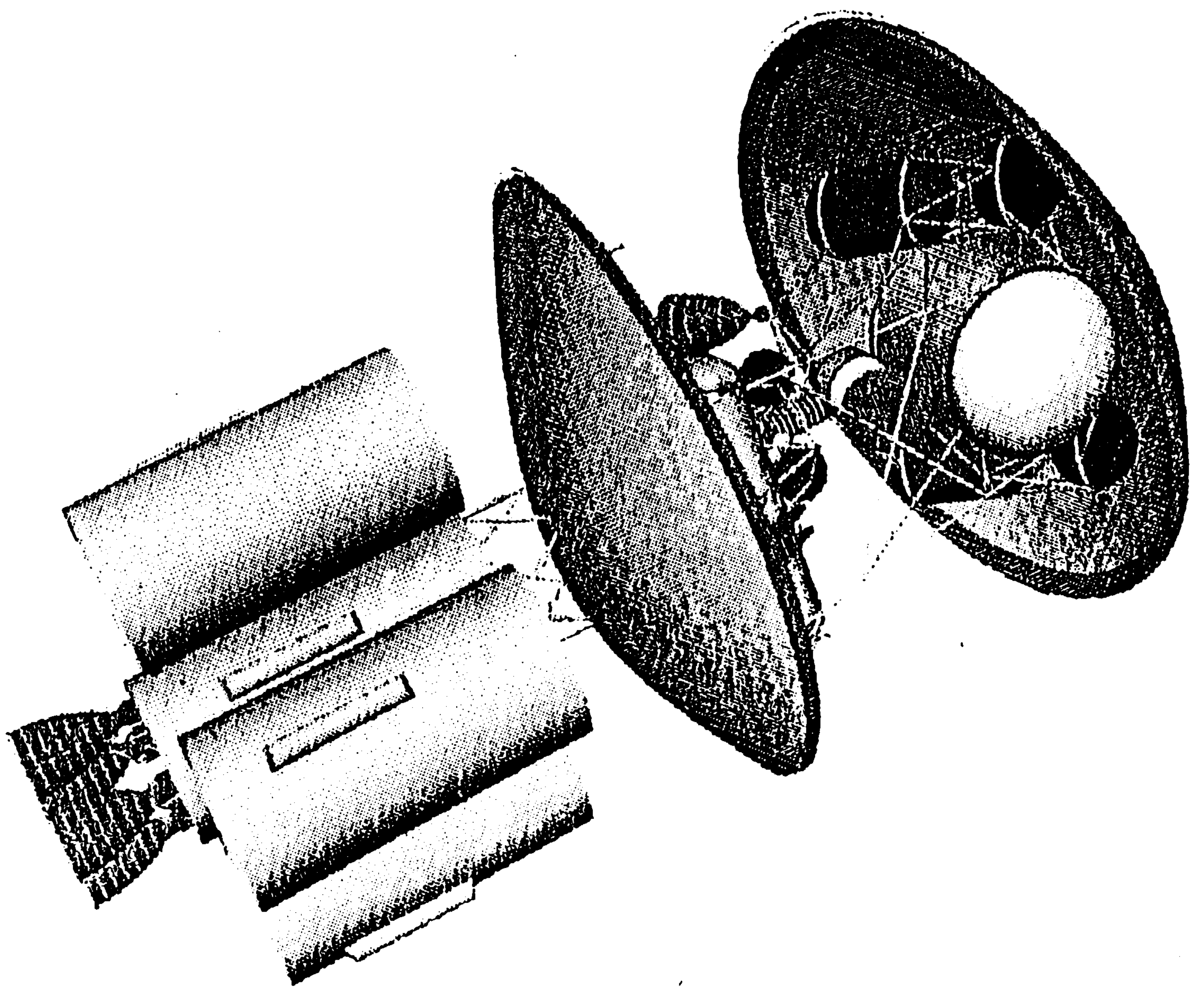
Cryo / Aerobraking
3D CAD MODEL



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Reference

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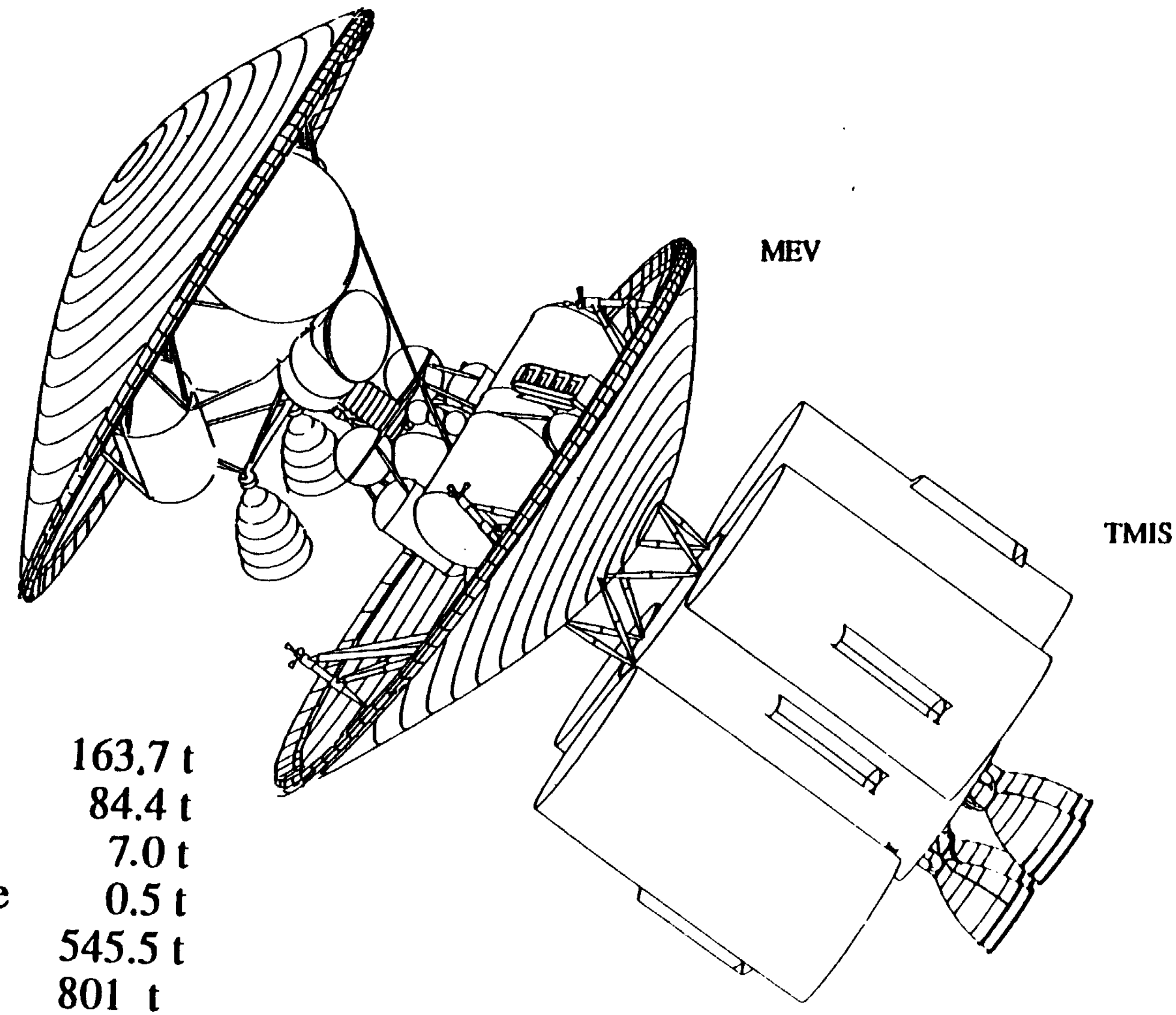
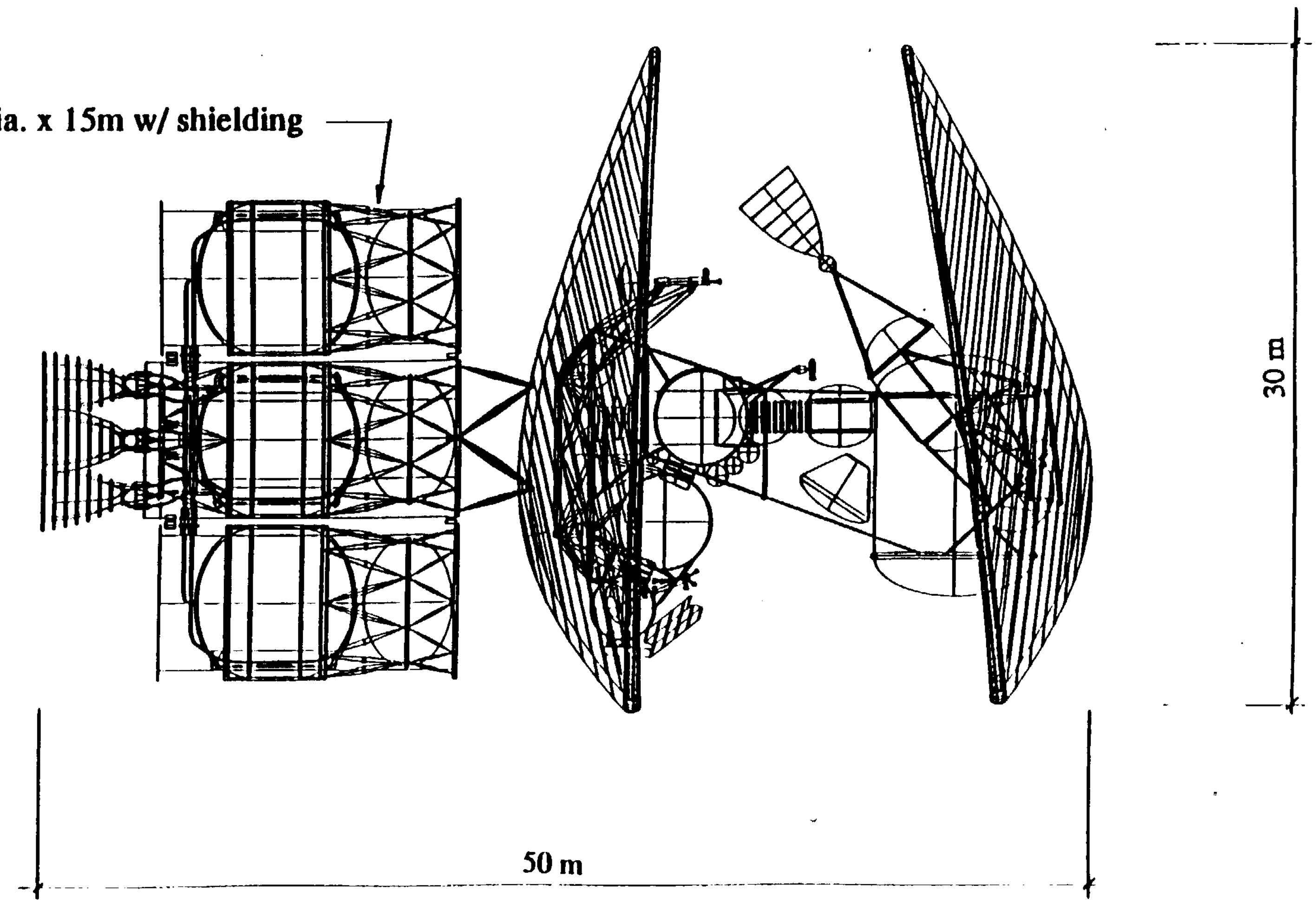
Cryo/AB Reference Configuration



MTV total	163.7 t
MEV total	84.4 t
ECCV	7.0 t
Interstage Structure	0.5 t
TMI stage total	545.5 t
IMLEO	801 t

Cryo/AB Reference Configuration



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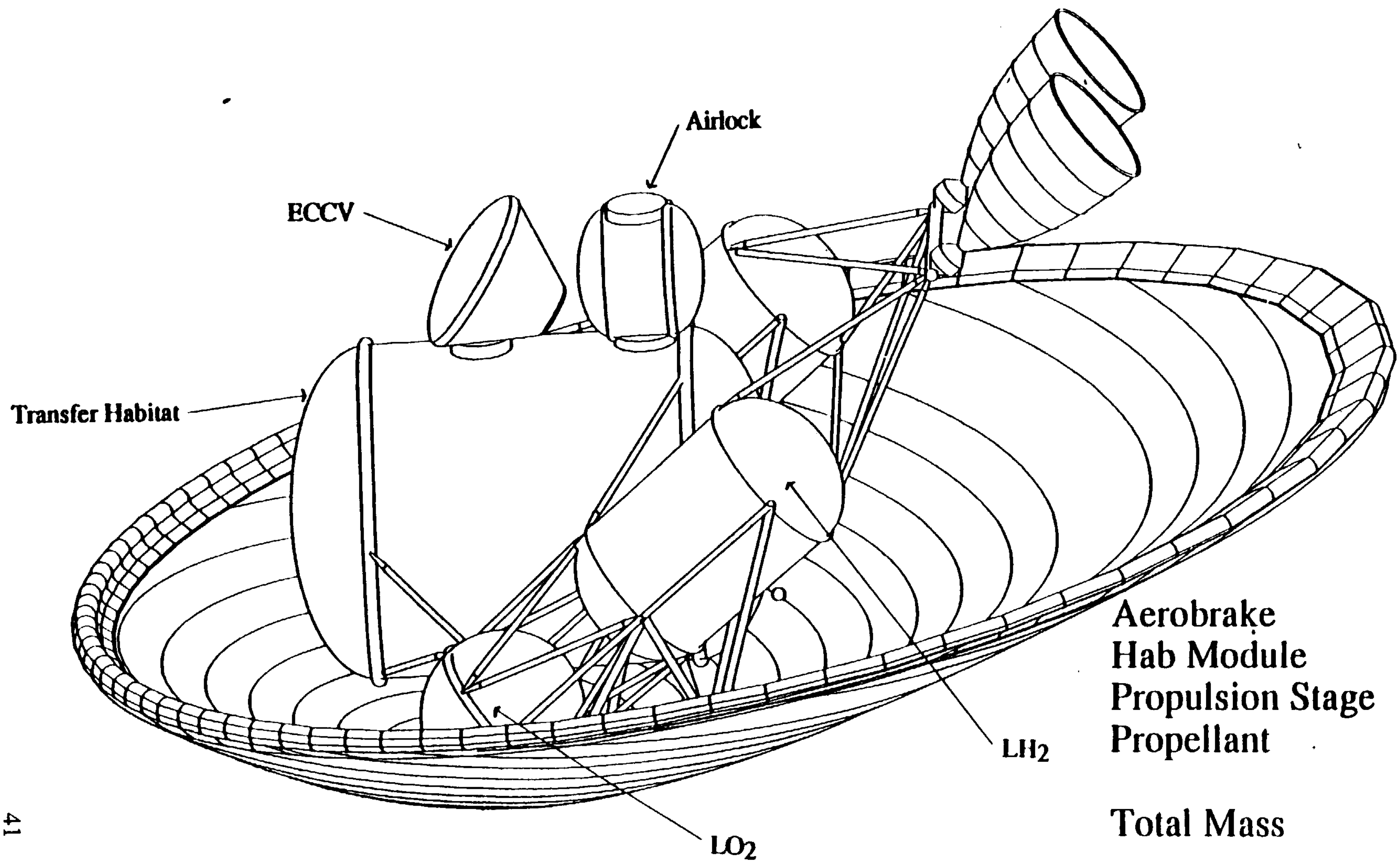
MTV Reference Configuration

The facing page illustrates the reference MTV / Aerobrake configuration. Shown are the transfer hab, ECCV, airlock and TEI propellant tanks and engines. The tank and engine structure is configured as shown to allow docking access by the MEV, and to allow the assembly to remain within the protected wake region.

MTV Reference Configuration



Aerobrake	23.8 t
Hab Module	36.6 t
Propulsion Stage	18.2 t
Propellant	85.1 t
Total Mass	163.7 t

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Options / Alternatives

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Options and Alternative Configurations

Alternative Landers

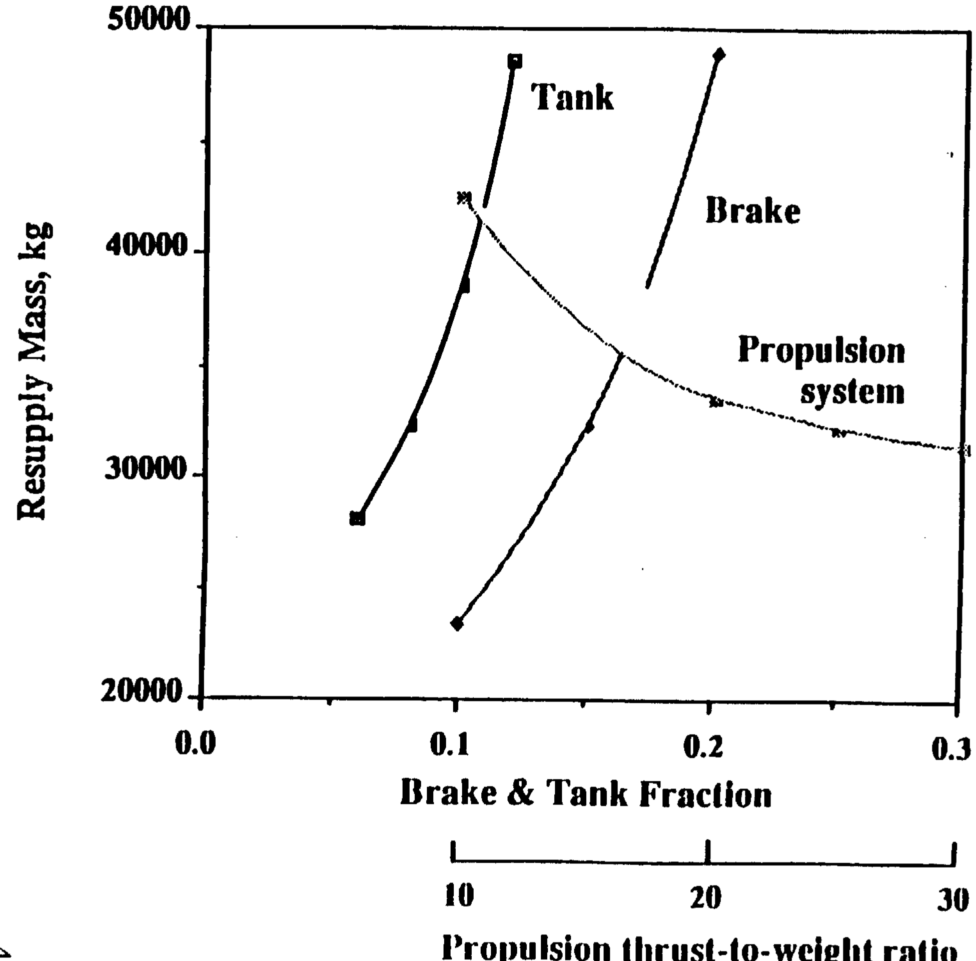
As an alternative to using the 0.5 L/D hyperboloid shaped aerobrake for a landing vehicle, investigations were made using a high (1.0+) L/D lifting body aerobrake shape and a Bi-conic shape. Both of these shapes extend the crossrange capability and are candidates for a reusable Mars Excursion Vehicle (RMEV), the criteria for which is given on the next pages. It appears that the high L/D aerobrake will be better suited for a reusable system, with fewer specialized parts. The Bi-conic will impose some restrictions on cargo that high L/D aerobrake will not, such as the delivery of a 10 crew habitat to the surface. In the case of the Bi-conic the habitat would have to be either specially built to fit the available space or the entire fleet of habitats would be scared to have this shape, at additional cost of fabrication. Other constraints became evident, while the high L/D aerobrake has limited visibility of the ground during landing operations, the Bi-conic has none.

Alternate Mission Vehicles

An all-propulsive cryogenic (chemical) vehicle was evaluated for near term conjunction missions. Conjunction missions were chosen due to the significant increase in IMLEO to mount an opposition mission using an all-propulsive vehicle (~ 1600 t IMLEO), while a conjunction mission would be in the range of the cryo/aerobrake mission under consideration (625-720t IMLEO). The advantage of the all-propulsive vehicle is that it has a short development time and can be ready early. The disadvantages are the limitation to the conjunction-class missions (long stay times at MARS, total trip time is long) and only the ECCV is recoverable, all other sections of the vehicle are expended in operations.

Reusable MEV Status

Reusable MEV Parametrics



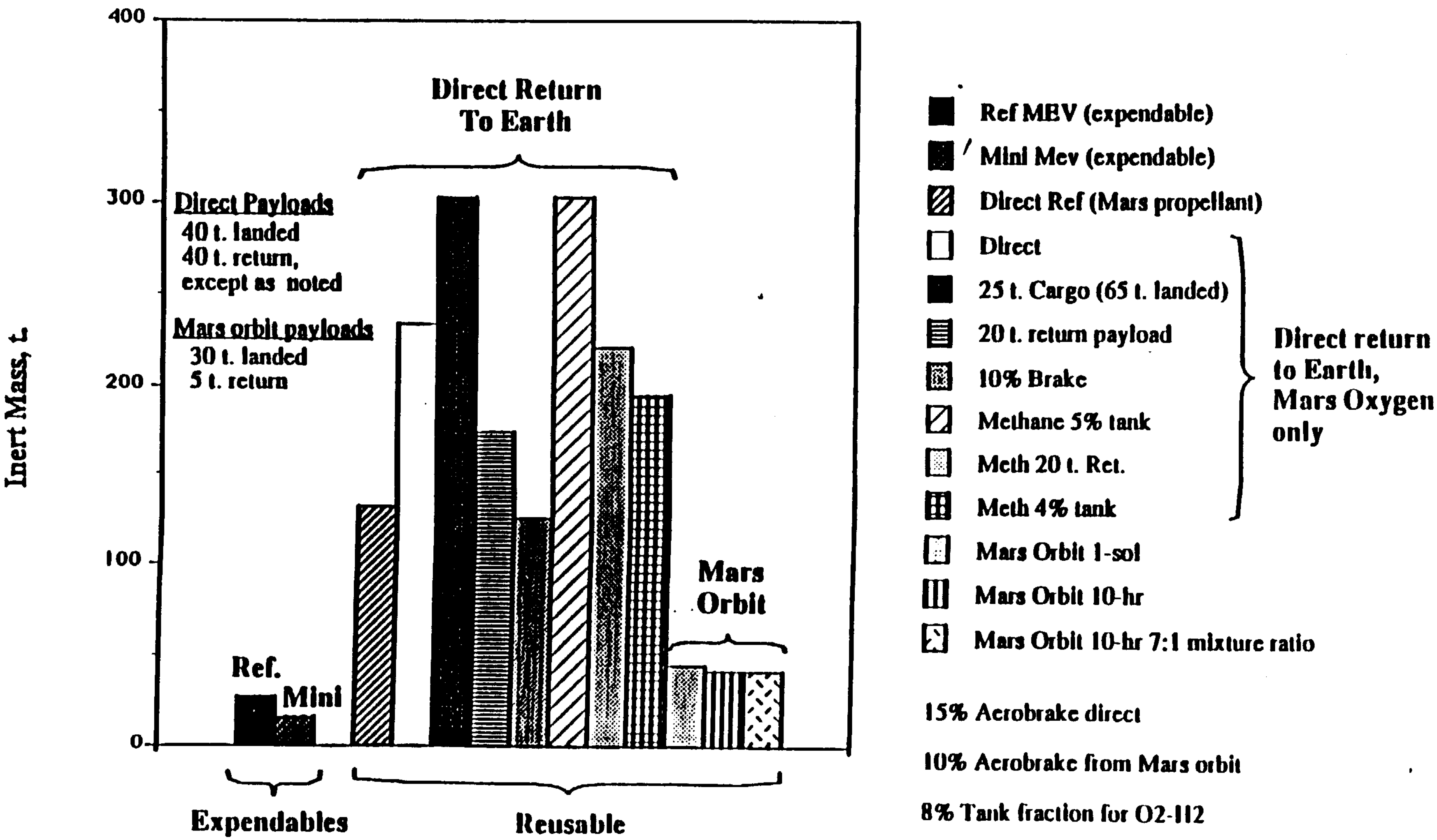
Assumptions

- Landing propellant and ascent H2 supplied in Mars orbit
- Ascent O2 from mars surface.
- 470 Isp
- Down payload 29.5 t. (25 t. + 4.5 t. c/m)
- Ascent payload 4.5 t. crew module
- Descent: aerobrake + 1200 m/sec
- Ascent ΔV 5500 m/sec
- Mixture ratio 6:1

Configuration Issues

- Location of payload at c.g. for landing
- Payload removal method
- Location of crew module for landing visibility
- Location of engines for landing and ascent
- Location and arrangement of tanks
- Landing gear design and placement for landing and ascent

Reusable MEV Sensitivities



15% Aerobrake direct

10% Aerobrake from Mars orbit

8% Tank fraction for O2-H2

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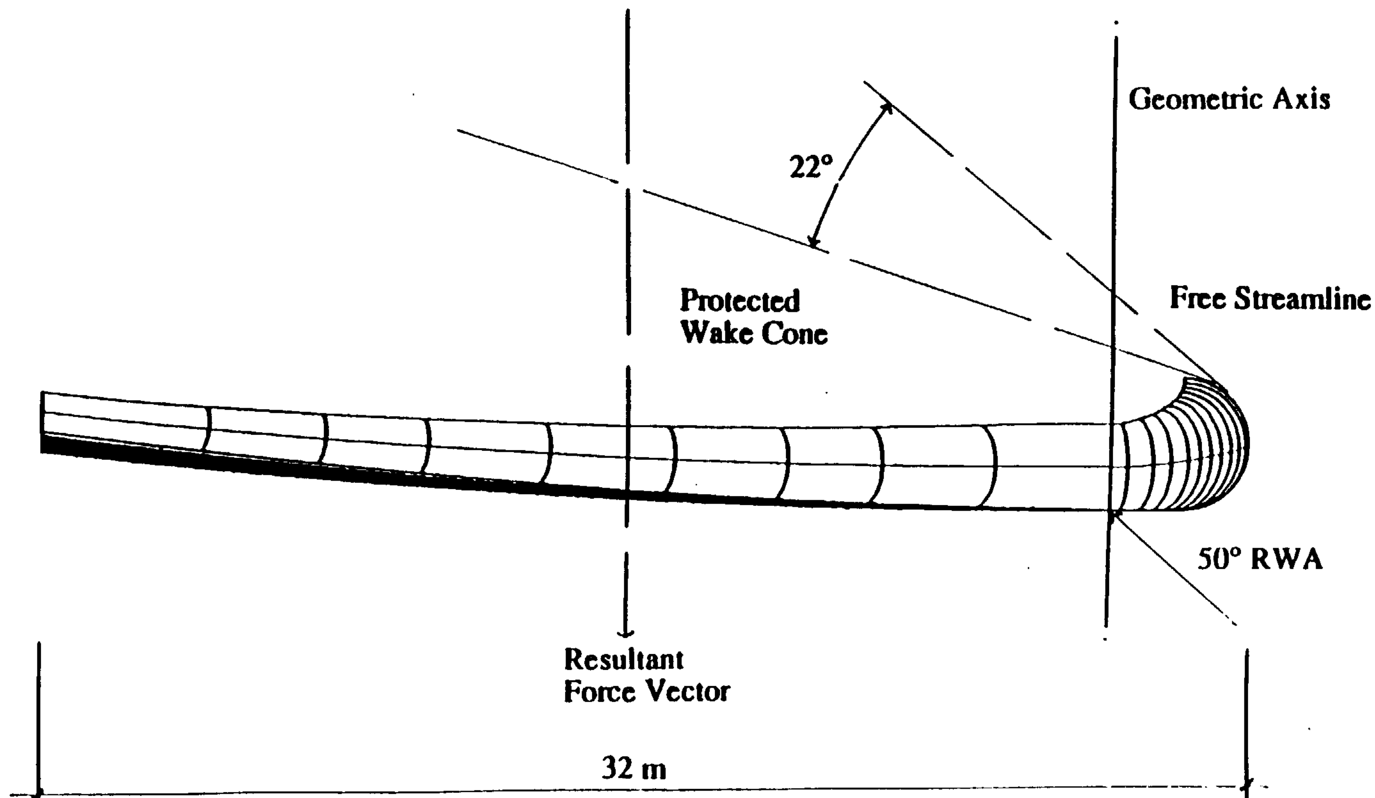
High L/D Aerobraking Constraints

Shown on the facing page are constraints as applied to the high L/D aerobrake, which were used to configure the reusable MEV. Aerobraking constraints include resultant force vector and protected wake cone, which impact the location of the MEV within the aerobrake.

High L/D (1.1) Aerobraking Constraints



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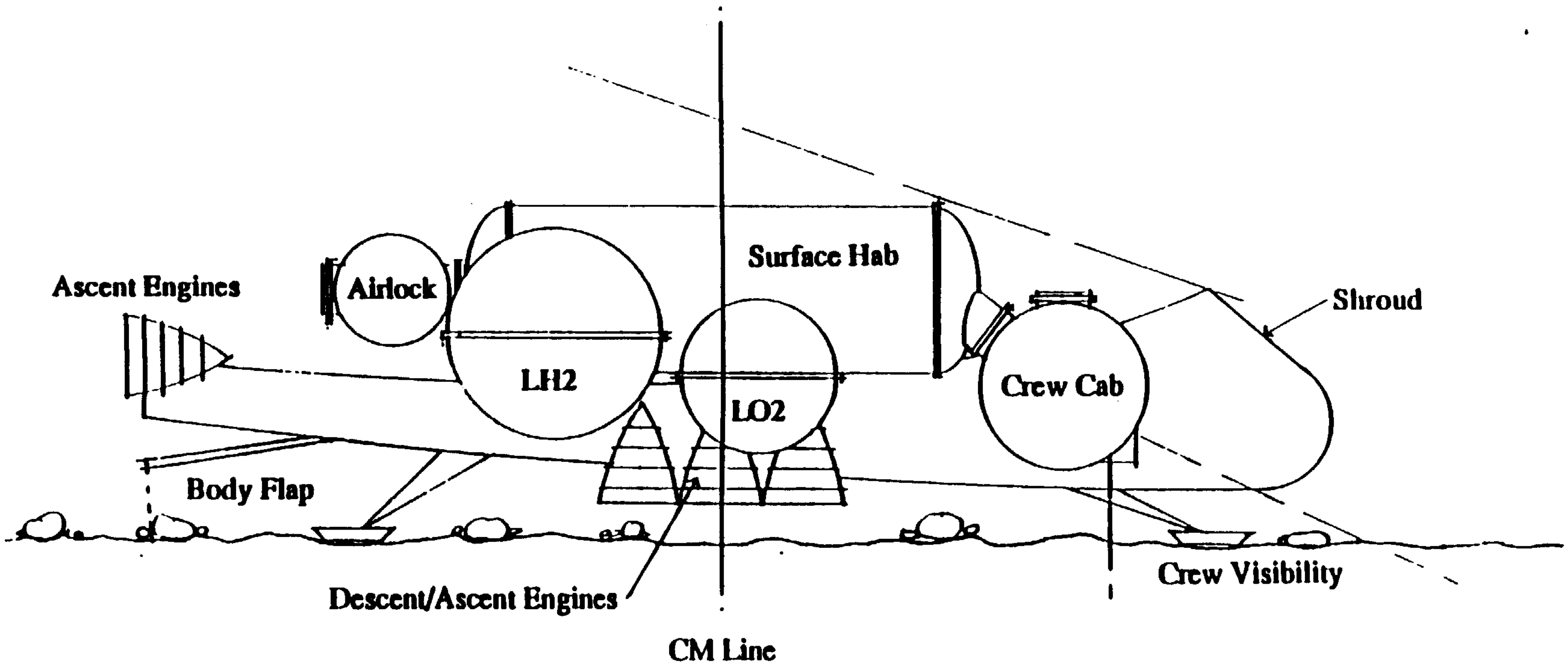
High L/D Reusable MEV Configuration

The high L/D reusable MEV is shown on the facing page. This configuration allows offloading of payload (shown here as a 4.4 m dia. hab module)by way of a ramp and track system, located at the rear of the vehicle. The ramp also acts as a body flap for aero-maneuvering. The crew cab is positioned so that the pilot has down and forward visibility until the landing area is selected. As the vehicle rotates into landing attitude, crew visibility will be limited to the surface directly below.

High L/D (1.1) MEV Configuration

- Payload at CM and removed out back of aerobrake via track system. Back landing legs retract to lower body flap to surface for payload removal
- Crew visibility for landing accommodated through front landing leg door
- Vehicle mass balanced to allow for flight with or without payload

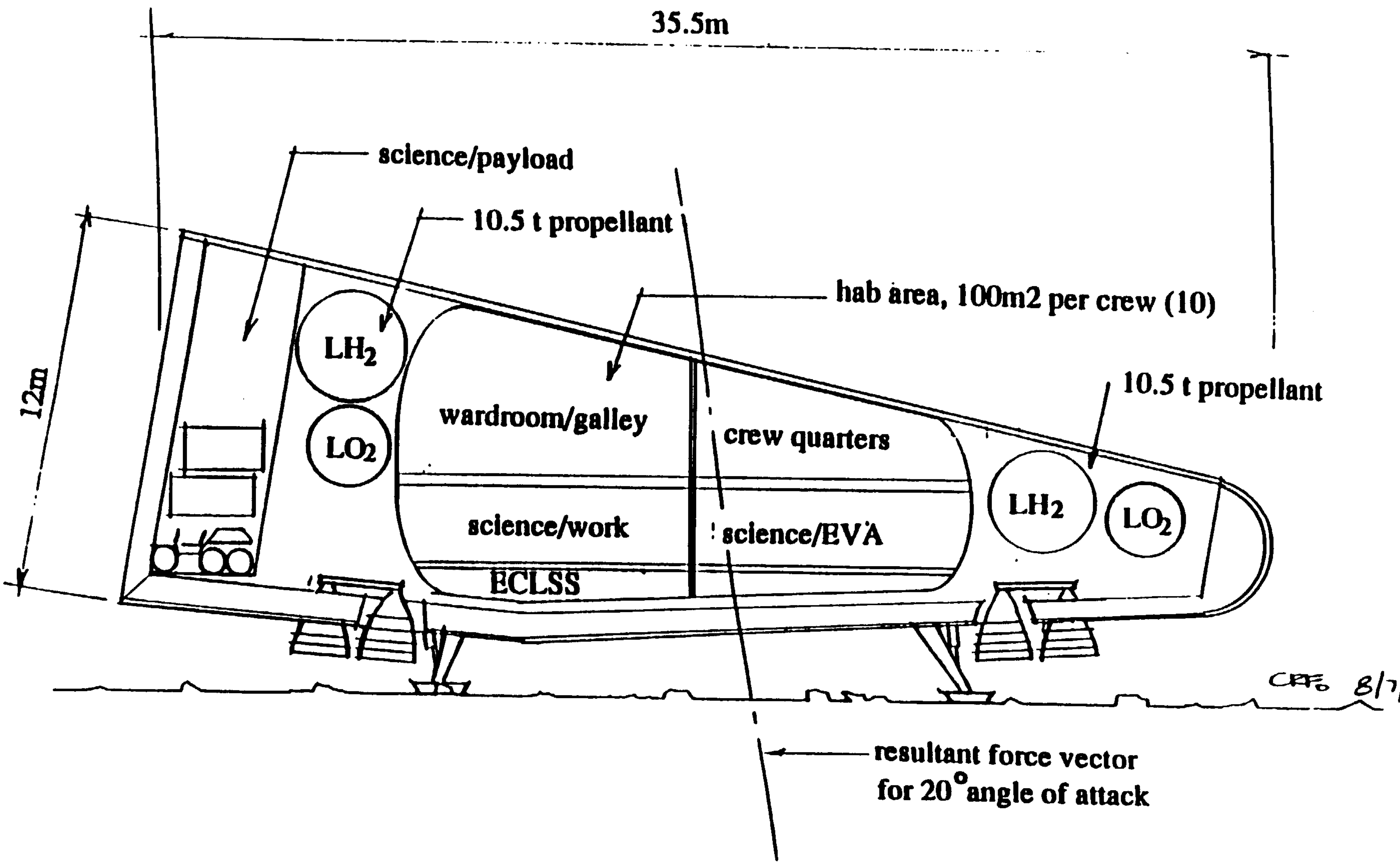


Bi-Conic Lander/Habitat

The Bi-conic lander/habitat is configured to be launched atop a 12m dia. HLLV, and delivered to Mars orbit via a separately launched TMI stage. The Bi-conic shape provides an L/D of 1.3 at a 20 degree angle of attack, and lands using a 6 engine configuration, split 3 forward and 3 aft. The unmanned vehicle is used in an expendable mode, and requires 21 metric tons of propellant for landing.

The 10 crew habitat module delivered to the surface is integrated within the bi-conic, and would not need heavy transportation equipment for deployment. The hab module weighs 40 metric tons when landed, and would need to be outfitted on the surface.

Bi-Conic Lander / Habitat



All-Propulsive Cryogenic Vehicle

Trades and Rationale

- Addition of MOI/TEI stage eliminates the need for a high energy aerocapture at MARS
- ECCV return for crew eliminates the need for a high energy aerocapture at Earth
- ECCV return is direct entry (Apollo- style, done before)

Mission Modes and Operations

- Vehicle assembled in SSF orbit.
- TMIS discarded after TMI burn.
- MOI burn and capture prior to MEV / aerobrake entry
- Aerobrake separates from MEV prior to landing.
- Crew cab ascent after surface mission, leaving lander, surface habitat
- Crew cab left in Mars orbit after rendezvous, docking and crew transfer.
- TEI burn.
- Crew return to Earth via ECCV

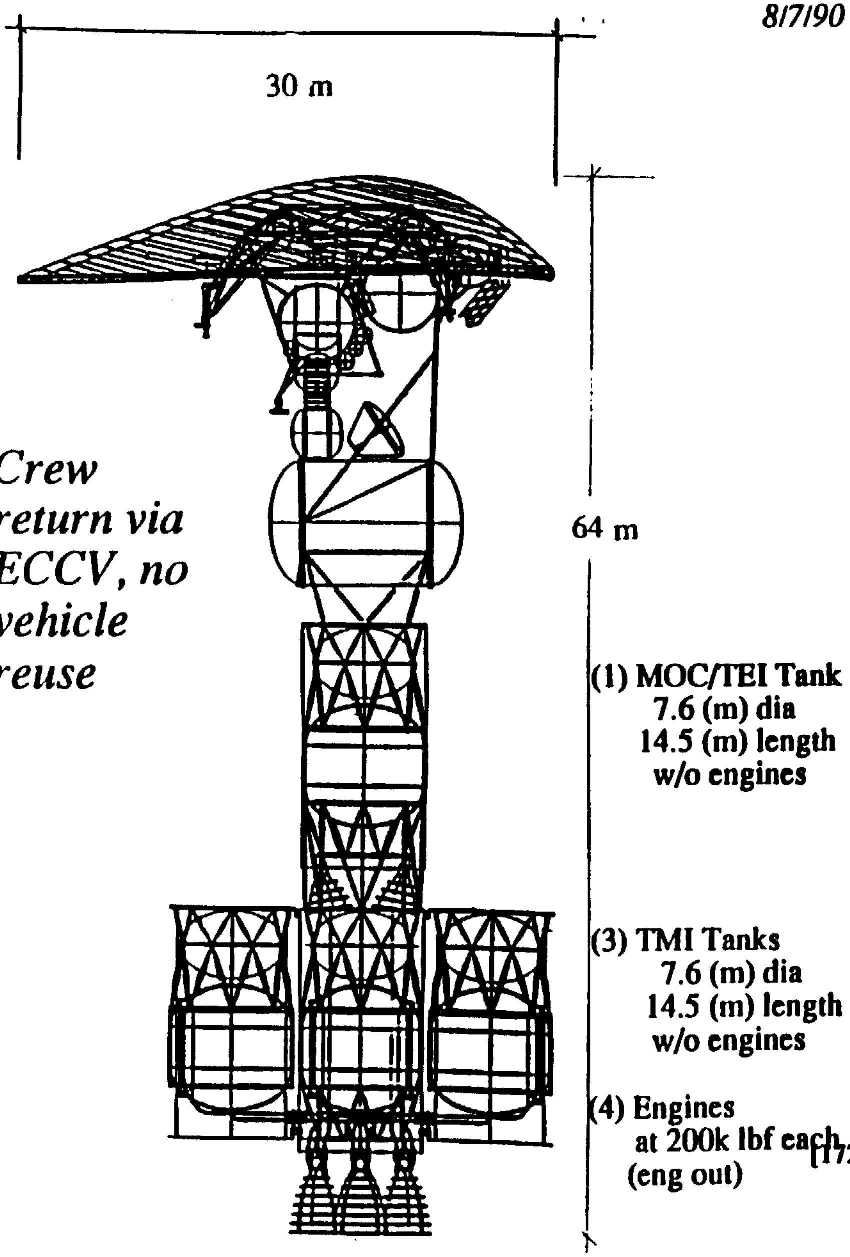
All Propulsive Chemical Vehicle for Conj Mission

Single MEV ; 5t surf cargo, crew of 4, Common tank sets for MTV stgs

dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV eng Isp=460

8/7/90

Crew return via ECCV, no vehicle reuse

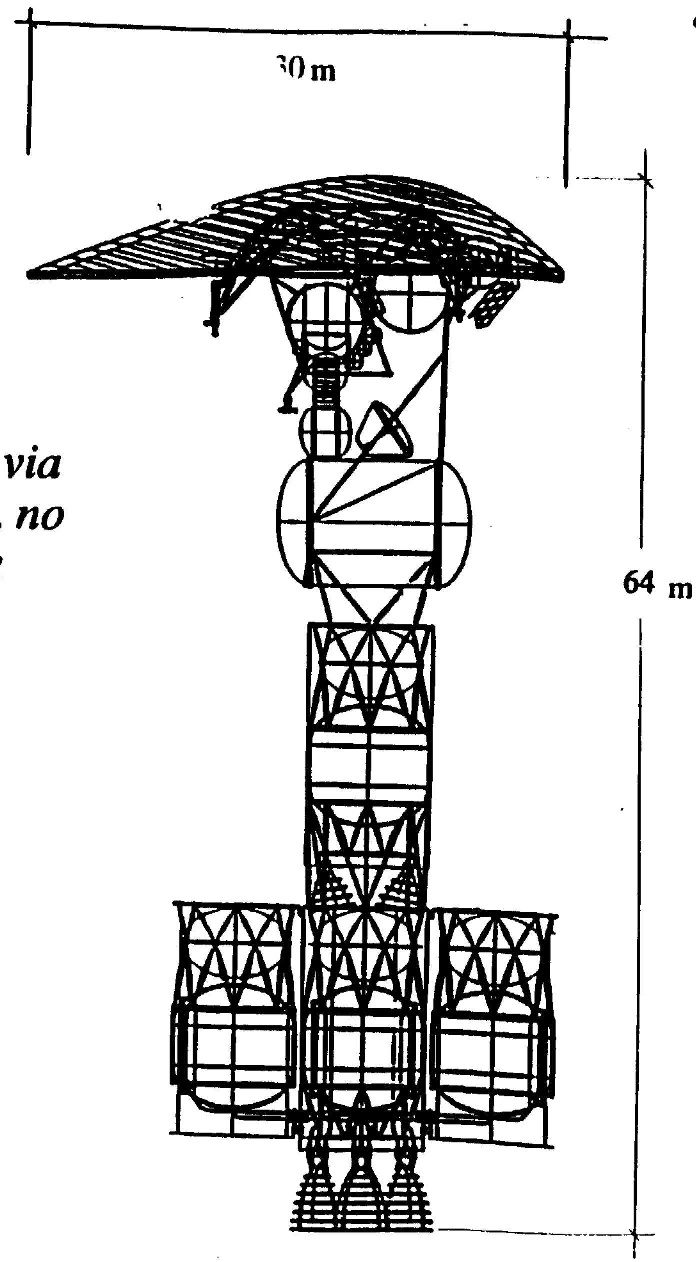


	Element	*Mass, kg*
[378]	MTV crew hab module 'dry'	39000
[398+371]	MTV hab consumables & resupply	14000
[179]	MTV hab mod science	0
sum	***MTV crew hab module***	**53000**
[128]	TEI usable propellant	15545
[551]	TEI outbound boiloff	1446
[545+546]	TEI inorbit boiloff	2698
sum	***Total TEI propellant***	**19689**
[541]	MOC usable propellant	62474
[538+539]	MOC outbound boiloff	3420
sum	***Total MOC propellant***	**66894**
sum	***EOC propel (Lunar case: returnto LEO)***	**0**
[118]	RCS propellant	842
[121]	Outb midcourse correction prop	1971
[122]	Inb midcourse correction prop	656
[161]	MOC/TEI propul stg inert	15510
sum	***MTV propulsion stg total***	**105562**
[i313]	MEV descent only aerobrake	6000
[63]	***MEV ascent stage***	*37406*
	Propellant / Isp	*Storable/340*
	MEV descent stage	*17019*
	Propellant / Isp	*Cryo/475*
[i66]	MEV surface cargo (3 crew for 90 days)	5000
[106]	***MEV total***	**65425**
[230]	***ECCV for crew return to LEO***	7000
[172-173+547]	TMI inert stage wt	39770
[173]	TMI propellant load	353360
[172]	***TMI stage total***	**393130**
[171]	***IMLEO** (all masses in kg)*	**624117**

A.1 Propulsive Chemical Vehicle for 2010 Conjunction Mission

Single 73t MEV carries 25t to surf, Common tank set for MOC/TEI stg
dV's: TMI dV= 4570 m/s, MOC=1160, TEI= 1186, Adv space eng's: Isp=475

Crew return via ECCV, no vehicle reuse

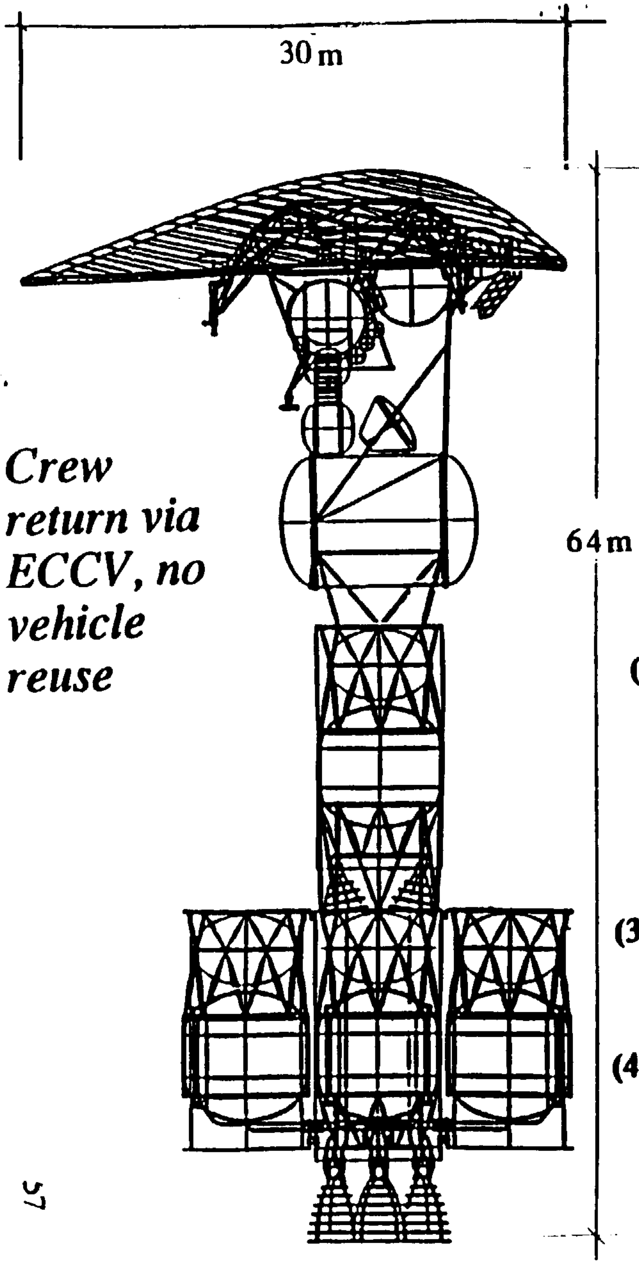


8/9/90	Element	Mass, kg
[378]	MTV crew hab module 'dry'	28531
[398+371]	MTV hab consumables & resupply	10560
[179]	MTV hab mod science	1000
[381]	***MTV crew hab module***	**40091**
[128]	TEI usable propellant	17677
[551]	TEI outbound boiloff	1415
[545+546]	TEI inorbit boiloff	2263
sum	***Total TEI propellant***	**21355**
[541]	MOC usable propellant	43602
[538+539]	MOC outbound boiloff	2468
sum	***Total MOC propellant***	**46070**
sum	***EOC propel***	n/a
[118]	RCS propellant	670
[121]	Outb midcourse correction prop	1744
[122]	Inb midcourse correction prop	521
[161]	MOC/TEI propul stg inert	12944
sum	***MTV propulsion stg total***	**83304**
[i313]	MEV descent only aerobrake	7000
[63]	***MEV ascent stage***	*22464*
	Propellant / Isp	*Cryo/475*
	MEV descent stage	*18659*
	Propellant / Isp	*Cryo/475*
[i66]	MEV surface cargo (4 crew for 30 days)	25000
[106]	***MEV total***	**73118**
[230]	*ECCV for crew return to LEO*	7000
	TMI inert stage wt	46870
[173]	TMI propellant load	417300
[172]	***TMI stage total***	**464170**
[171]	***IMLEO***	[illegible]

All Propulsive Chemical ...

Single MEV ;30t surf cargo, crew of 3 for 90 days on surf, 1 crew member left in orbit , Common tank sets for MTV stgs
dV's: TMI dV= 3900 m/s, MOC=1530 m/s, TEI= 860, E arr Vinf=3200,TMI, MOC,TEI eng Isp=475, MEV eng Isp=460

Crew return via ECCV, no vehicle reuse



(1) MOC/TEI Tank
7.6 (m) dia
17.0 (m) length

(3) TMI Tanks
7.6 (m) dia
17.0 (m) length

(4) Engines
at 200k lbf each
(eng out)

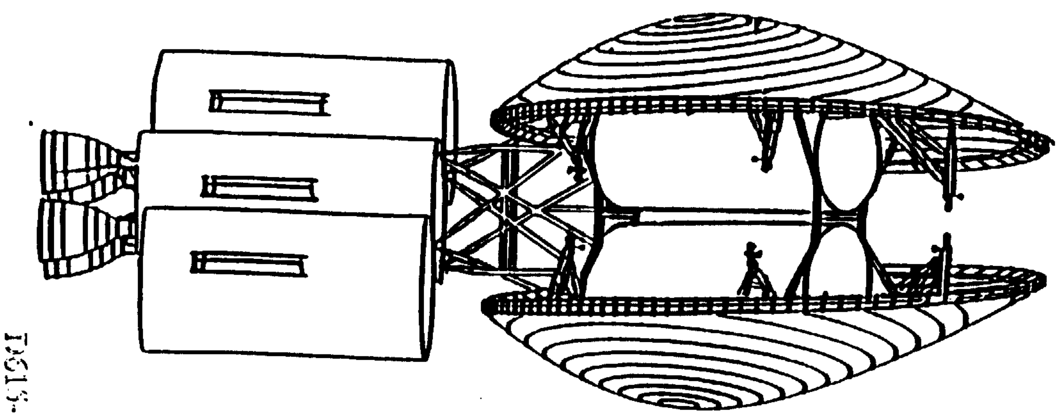
Revision 3 8/7/90

	Element	*Mars*	*Lunar* offloaded TMI
[378]	MTV crew hab module 'dry'	39000	39000
[398+371]	MTV hab consumables & resupply	14000	14000
[179]	MTV hab mod science	0	0
sum	***MTV crew hab module***	53000	53000
[128]	TEI usable propellant	15713	
[551]	TEI outbound boiloff	1455	
[545+546]	TEI inorbit boiloff	2716	
sum	***Total TEI propellant***	19884	27471
[541]	MOC usable propellant	72214	
[538+539]	MOC outbound boiloff	3752	
sum	***Total MOC propellant***	78910	48015
sum	***EOC propel (Lunar case: returnto LEO)***	0	64056
[118]	RCS propellant	852	917
[121]	Outb midcourse correction prop	2274	0
[122]	Inb midcourse correction prop	663	0
[161]	MOC/TEI propul stg inert	16335	16335
sum	***MTV propulsion stg total***	115974	156729
[1313]	MEV descent only aerobrake	12000	12000
[63]	***MEV ascent stage***	*24262*	*11716*
	Propellant / Isp	*Cryo/460*	*Cryo/460*
	MEV descent stage	*23190*	*22739*
	Propellant / Isp	*Cryo/460*	*Cryo/460*
[166]	MEV surface cargo (3 crew for 90 days)	30000	30000
[106]	***MEV total***	89452	76455
[230]	***ECCV for crew return to LEO***	7000	0
[172-173+547]	TMI inert stage wt	45800	45800
[173]	TMI propellant load	407680	315199
[172]	***TMI stage total***	453480	360999
[171]	***IMLEO (all masses in kg)***	718910	647183

Cargo Chem/aerobrake Veh for one way 2018 Conjunction Mission

Unmanned, 2 cargo landers (46.5 t surf cargo each), 10 t navigation set, no MTV propulsion stg, TMI stg Isp=475

Revision 2 7/30/90



Element	*mass (kg)*
MTV Mars aerobrake	0
MTV crew hab module 'dry'	0
MTV consumables & resupply	0
MTV science	0
MTV propulsion stage	0
MTV propellant load	0
MTV total	0
MEV Mars orbit capture & desc aerobrake	15138
MEV descent stage	21457
MEV surface cargo	46457
MEV total	84349
x 2	168968
ECCV	0
Cargo to Mars orbit only (navigation set)	10000
TMI inert stage wt	25770
TMI propellant load	231920
TMI stage total	257690
IMLEO	436658

Mac chart: Cargo chem/ab 2018 wt cover pg
Veh synthesis model run #: marschemmtv.dat;37

D615-10026-2

Common Mars/Lunar Lander Vehicle - Cargo & Manned Versions

Mars desc propul dV: 773, Asc dv:5319, Lunar asc & desc dV: 2100, all cryo prop Isp=475

***Single stage vehicle** - aeroshell , cargo and landing legs left on surface*

Element	Mars Cargo (desc only)	Mars *Manned (single stg desc/asc veh)	Lunar Cargo (desc only)	Lunar *Manned (single stg)
Ascent cab	0	3500	0	3500
Stg inerts	5374	5374	5374	5374
Aeroshell	7500	7500	n/a	n/a
Surf Cargo	30000	700	30000	**12612
Asc prop	n/a	16082	n/a	5310
Desc prop	7900	5255	20658	16027
RCS prop	893	1341	893	1341
Total wt	**51668**	**39752**	**56925**	**44164**

* *Manned: crew of 3 or 4 for very short surf stay time (a week or less)*

** *Maxium surface cargo load for manned lunar case when all tanks are full*

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Architecture Matrix

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Reference Matrix to Alternative Architectures

In considering a complex task, it is useful to organize it into a heirarchy of levels. The higher levels are more important or more encompassings, while the lower levels include more detail or are more specific. Constraints (e.g., requirements and schedules) flow down from the higher levels and solutions or implementations build up from the lower levels. The first figure shows a heirarchy of six levels from national goals to performing subsystems. The following section discusses the fourth level, exploration architectures, in terms of the lower levels: element concepts and performing subsystems. Selection of preferred architectures will require the Government (the National Space Council, the President, and the Congress) to first define the top three levels.

Implementation Architectures

Seven architectures have been selected for examination: four different propulsion types (Cryogenic/Aerobrake, NEP, SEP, and NTR): two variations of In-Situ Resource Utilization (ISRU) for propellants with Cryogenic/Aerobrake propulsion (Lagrange point 2 refueling and Mars surface refueling); and a cycling spacecraft concept. Three basic levels of program scope are identified: small, moderate, and ambitious.

Multiple options can be generated within the basic architectures, varying launch vehicle capacity, orbital node type, and mission profile and propulsion type for the various Lunar and Mars vehicles.

Aerobraking is found to be applicable to all seven architectures, placing it as a 'critical' technology. Electric propulsion leads to the lowest reference vehicle mass, and also almost the lowest resupply mass. ISRU/Cryo leads to the lowest estimated resupply mass since most of the propellant is derived locally rather than coming from Earth.

Cost Models

Cost estimation is being performed using "parametric" methods. This technique uses a parameter, usually weight, as an input to empirically derived equations that relate the parameter to cost. It should be recognized that the source data for the cost models is past program experience, while the hardware being estimated will be built one or two decades from now. Therefore these cost estimates should be assumed to have a standard deviation on the order of +-100%. Hardware at technology readiness level 5 may be assumed to have a standard deviation in cost estimate of +-30%. No revenues from sale of products, services, or rights (i.e. patent rights, data rights), or commercial investment, are assumed in the cost estimates. These might appear in a scenario such as the Energy Enterprise.

Aa an example, the cost estimate for a NEP architecture shows an average annual funding level of $8 billion per year after initial ramp-up.

The principal cost drivers identified include number of development projects, reuseability, mass in Earth orbit, and mission/operational flexibility.

Analysis Methods

Individual trade studies are performed within each architecture to optimize it against evaluation criteria. The principal evaluation criteria to date has been initial mass in low Earth orbit, as a proxy for cost. The results of this optimization will then be compared to each other in groups. The early Mars group will compare all-propulsive, aerobraking,

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direct travel, and nuclear thermal among themselves. The electric propulsion group will compare SEP and NEP. The innovative group will compare Lunar oxygen to cycler orbits. These concepts may both be retained if it is advantageous to do so. Finally, the choice between early Mars and Late/Evolving Mars will need to be made on the basis of cost, risk, and performance, while combining the best features from each group.

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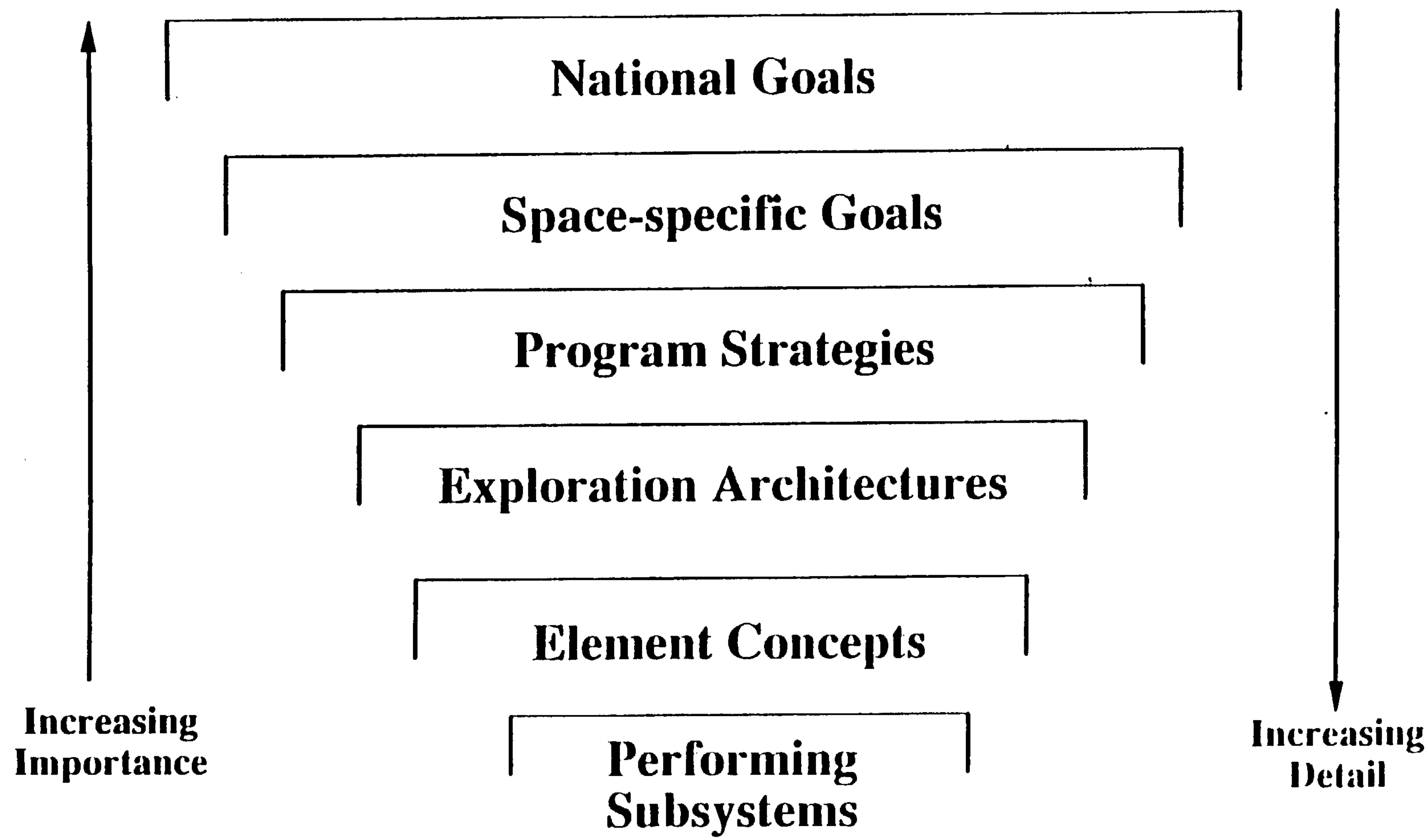
Logical Types for Space Programs

Architectural planning for a space program deals with many levels of information. A major space program like the space exploration initiative must respond directivity to national goals in traceable ways. While we do not determine national goals, it is our business to understand how exploration architectures can be evaluated in terms of national goals.

National goals translate to space specific goals for specific exploration programs such as science emphasis or expanding human presence. These in turn can lead to program strategies for space-specific goals such as low risk, high technology, low cost and so forth. Finally, exploration architectures are integrated assemblages of systems, mission profiles, and operations, necessary to satisfy program goals.

"Logical Types" for a Space Program



Each logical type subsumes all the subordinate types

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Overall Study Flow

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown on the facing page. Combinations of major technologies, such as electric propulsion and aerocapture, were quickly determined to be uneconomic in view of high development costs. Further, we found that electric propulsion systems could perform both crew and cargo Mars missions if crews are transported to and from the electric system at about lunar distance by a lunar transfer vehicle.

New systems introduced included nuclear thermal rocket (NTR) and Mars direct. NTR was introduced as an option by NASA during the "90-day study". We introduced the Mars direct profile (everything is landed on Mars; the return propulsion system is loaded with oxygen and perhaps fuel as well on Mars) in March 1989. Martin-Marietta subsequently publicized one variant of this concept.

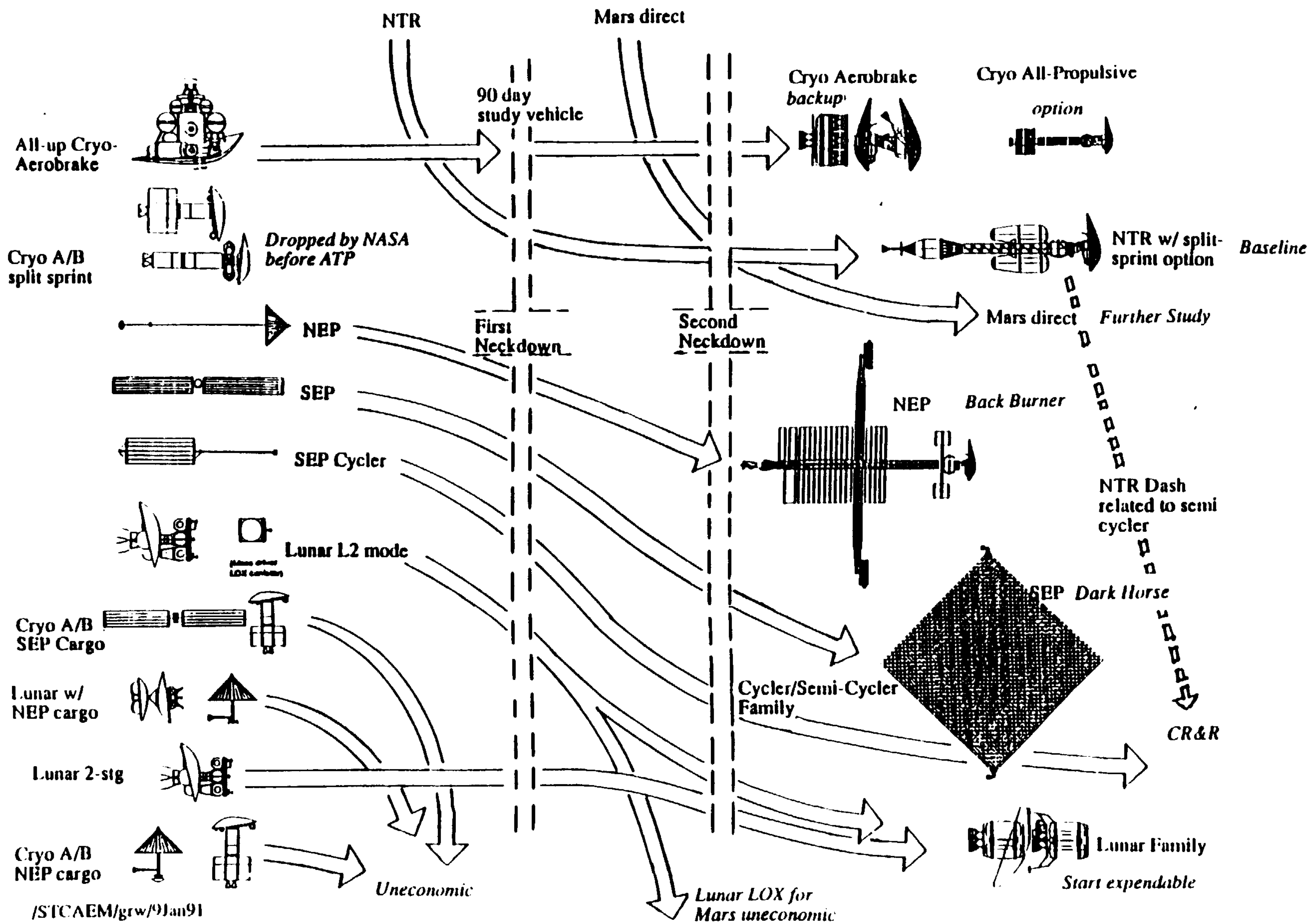
Lunar oxygen for Mars missions was found to be uneconomic because of long payback time for the launch mass required to emplace lunar oxygen production on the Moon. Lunar oxygen has a reasonable return on investment for lunar transportation at two or more lunar trips per year.

The cycler architecture was broadened to include semi-cyclers. Late in the study we introduced an NTR-dash mode (described later in this briefing) closely related to the semi-cyclers.

Overall Study Flow

ADVANCED CIVIL SPACE SYSTEMS

BOEING



Program Implementation Architectures

We have selected seven program implementation architectures for architectural analysis. These seven architectures incorporate the advanced propulsion options of principal interest in complete evolutionary architectural scenarios for lunar and Mars exploration. The facing page lists the features of each architecture and the rationale for selection of each.

Some of the architectures include suboptions. For example, the nuclear electric propulsion and solar electric propulsion architectures include optional use of the electric propulsion system for lunar cargo delivery from LEO to lunar orbit. The L2-based cryogenic aerobraking architecture includes use of NTR and NEP vehicles for LEO to L2 cargo delivery as options, and also includes a cryogenic all-propulsive conjunction mission option.

Program Implementation Architectures

Architecture	Features	Rationale
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	NASA 90-day study baseline
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.	High performance of nuclear electric propulsion
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.	High efficiency of solar electric propulsion; find cost crossover for array costs.
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.	High Isp of nuclear rocket enables avoidance of high-energy aerocapture at Mars.
L2 Based cryogenic/ aerobraking	L2-based operations; use of lunar oxygen.	L2 base gets out of LEO debris environment. Lunar oxygen reduces resupply by ~ factor 2.
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	Eliminates Mars orbit operations.
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	Eliminates boosting massive Mars transfer vehicle.

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SEI Program Scopes for Transportation Architecture Analysis

There are many space-specific goals and program strategies. We believe that transportation architectures will respond mainly to program scope. Some architectures are best suited to small program with early goals and others best suited to long range larger programs with ambitious goals. We have selected three representative scopes for small, moderate and large programs as illustrated on the facing page. These scopes permit definition of transportation requirements in terms of numbers of people and amounts of cargo transported to particular locations on particular schedules.

The second important feature of the scopes we intend to investigate is that they cover a scale factor greater than ten. A man tended science station may have few people on the Moon for short periods, or few people on Mars for short periods every other year. Permanent science bases will involve a dozen or so people. Industrial development of lunar resources on a scale of helium-3 scenarios leads to numbers of people presently estimated in the range of thousands by 2050. Beginnings of humans settlement of Mars involves numbers in the range hundreds to thousands. The 20-25 horizon for SEI is expected to permit growth in numbers of people only to dozens or so.

SEI Program Scopes for Transportation Architecture Analysis

Descriptor	Small	Moderate	Ambitious
Lunar Operations	Man-tended science station	Permanent science base 6 - 12 people	Industrial development of lunar resources
Mars Operations	Expeditionary visits ~4 people	Permanent science base 6 - 12 people	Beginnings of human settlement

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Three Activity Levels for Architecture Evaluation

We established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in-space transportation technologies as baselines for greater activity levels.

Activity levels were selected with underlying program objectives in mind:

(1) The minimum lunar program establishes astrophysical observatories on the Moon and provides a man-tending capability to maintain them. To the extent that man-tending lunar visits are not needed for the observatory system, the transportation capability can be used to explore interesting lunar sites for lunar geoscience objectives.

(2) The minimum Mars program is very similar to Apollo, i.e. six sites visited for short periods (two sites per mission and three missions); samples obtained within a few km. of each landing site. If the manned visits are preceded by suitable robotic missions, the scientific payoff for these visits can be high relative to the investment.

(3) The "full science" lunar program adds human permanence at the Moon for extensive scientific and technological exploration. Where the minimum program offers very little opportunity for lunar geoscience, this program offer much. It also permits development of in-situ resource technology for production of surface systems. The reference program also emplaced a lunar oxygen production system to serve the transportation system.

(4) The "full science" Mars program multiplies by several the crew person-days on Mars by including more missions and by more staytime per mission. This program falls short of a permanently-occupied base on Mars, but achieves surface stays greater than a year.

(5) The lunar industrialization program adopts production of helium-3 as a strawman industrial objective and places enough facilities and infrastructure on the Moon by 2025 to return 1 GWe helium-3 fusion fuel to Earth.

(6) The Mars settlement program moves towards Mars settlement. A robust nuclear electric propulsion system is fielded, with convoy flights by 2015. Mars population reaches 24 by 2025, and the transportation system is capable of increasing Mars population by 24 per opportunity by 2025.

Three Activity Levels for Architecture Evaluation

Minimum	Median (full science)	Industrialization /settlement
Just enough to meet President's objectives	*Meet science objectives of lunar/Mars exploration*	*Return of practical benefits to Earth*
• Permanent lunar facilities, not permanent human presence	• Human permanence • Opportunity for lunar geoscience • In-situ resource technology	• Extensive facilities and infrastructure on the Moon by 2025 • Lunar population 30 by 2025
• Astrophysics observatories • Man-tending capability • Explore interesting sites		
• Three missions to Mars • Similar to Apollo • Two sites per mission • Samples within a few km. of landing sites	• Order of magnitude more crew time on Mars • Approaches permanent base (stay time >1 year)	• Mars population 24 by 2025 • Capable of increasing Mars population by 24 per opportunity by 2025.

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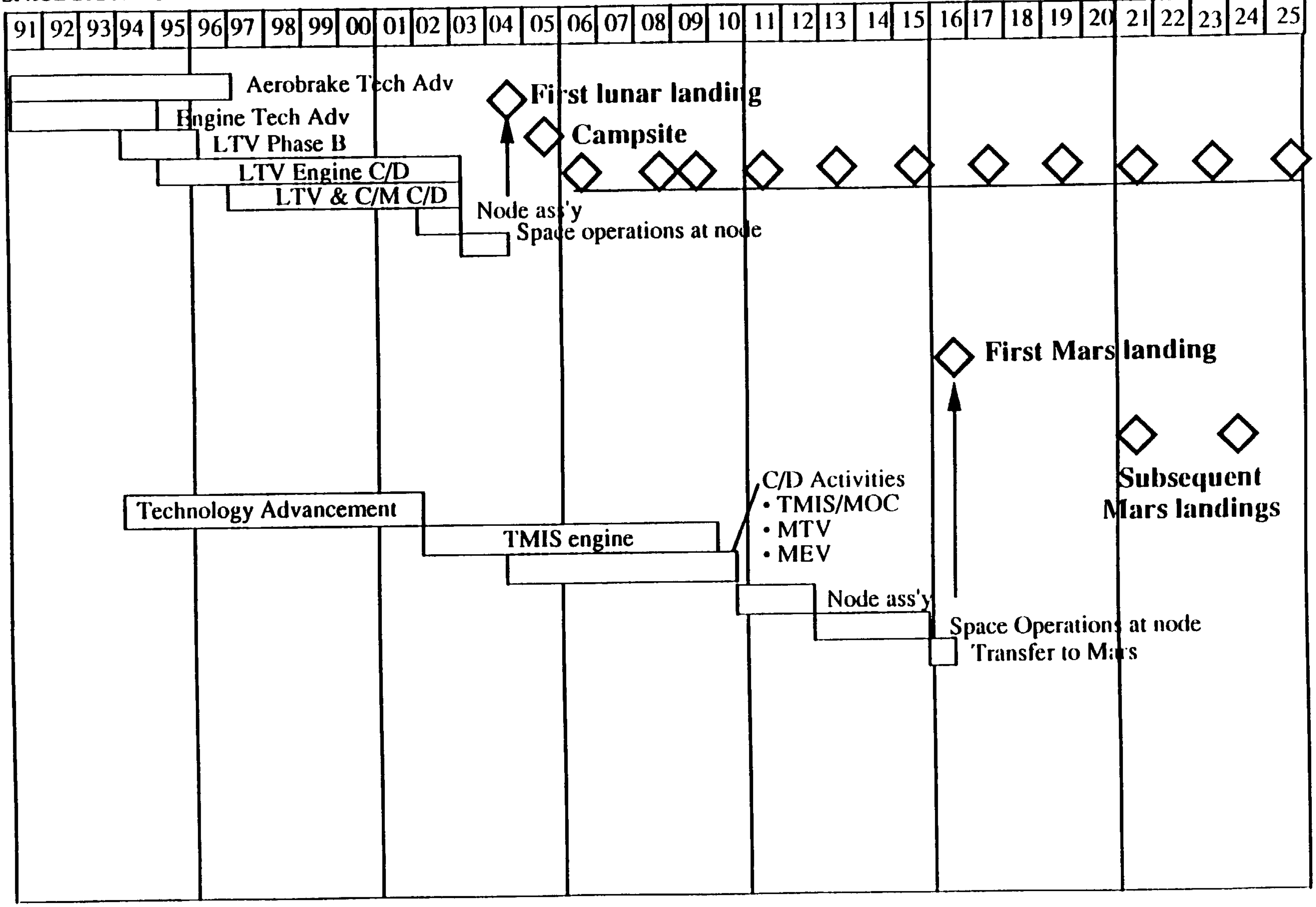
Minimum Program

The minimum program reference averages about 1/2 lunar trip per year and has only three Mars missions. Lunar science facilities are man-tended. Each Mars mission carries two landers (MEVs) for added exploration capability and a measure of rescue capability. Surface stays are about 30 days. Lunar and Mars in-space transportation systems are expendable.

Minimum Program



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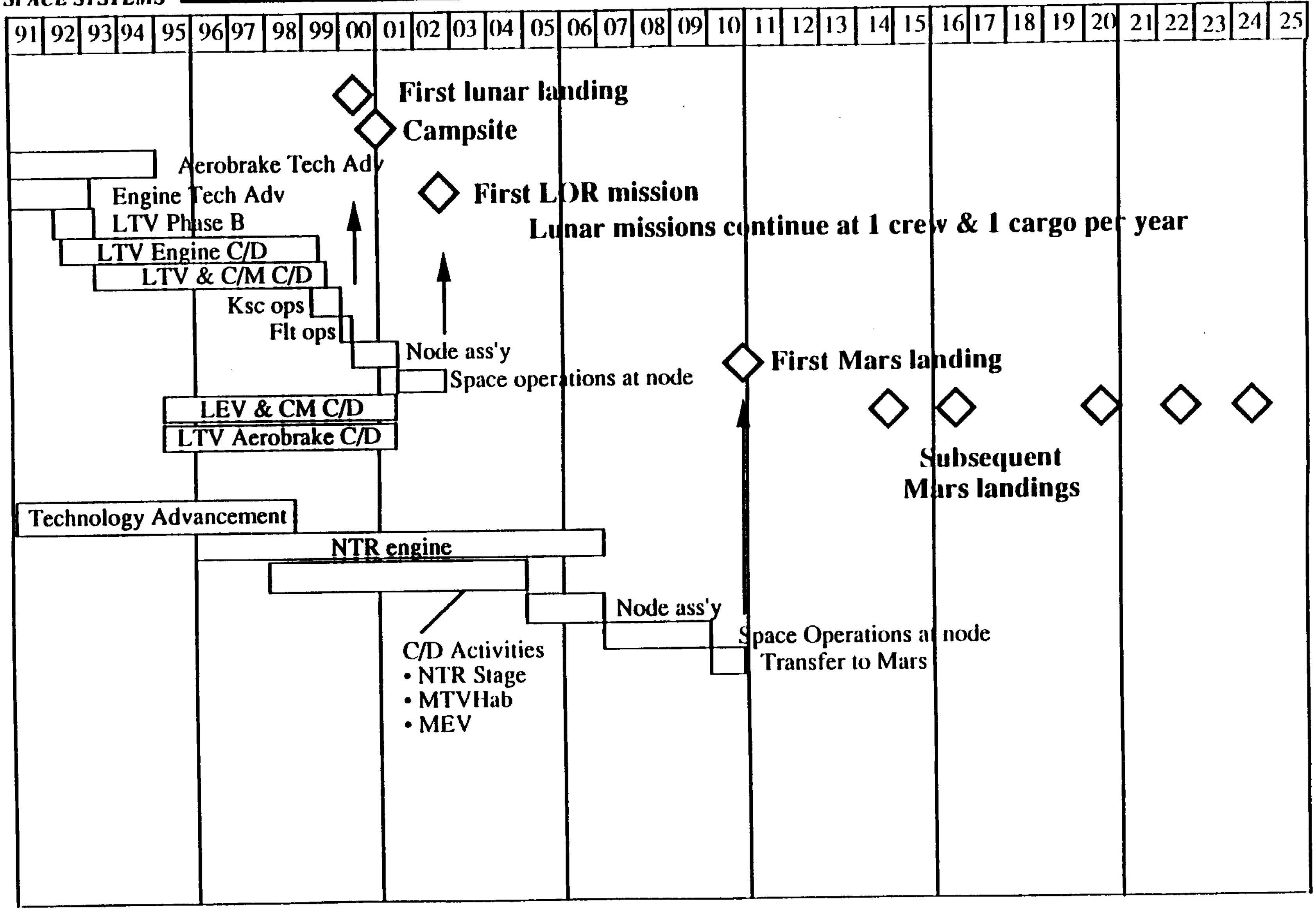
Full Science Program

The full science program reference has about 2 lunar missions per year, to establish permanent human presence on the Moon with adequate supplies and equipment for extensive science and exploration. Lunar oxygen for lunar transportation is introduced about mid-way through the lunar program. Six Mars missions are accomplished, with later missions staying on Mars for more than a year. The Mars missions use multiple landers, as many as four late in the program.

Full Science Program



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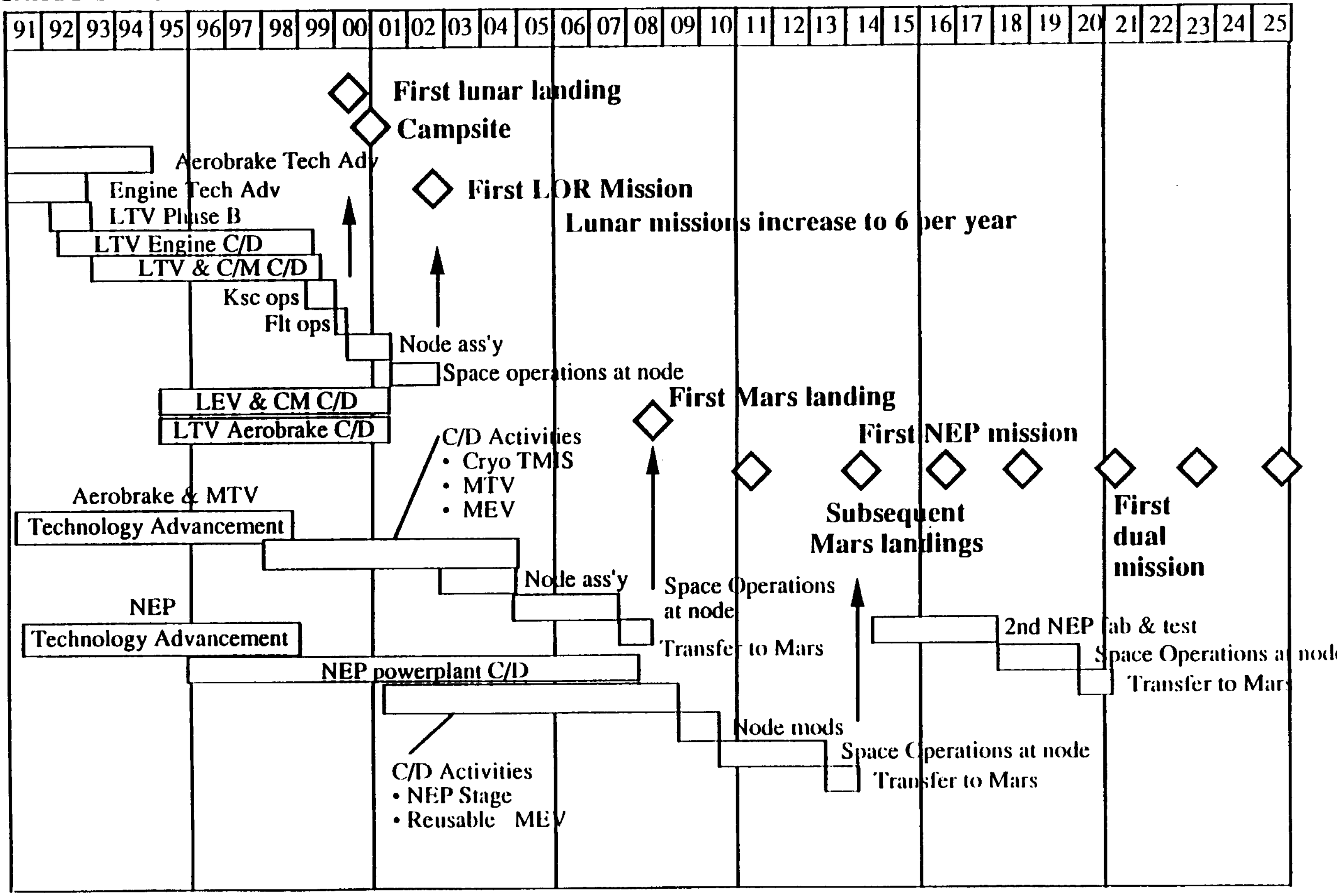
Industrialization and Settlement Program

The industrialization and settlement program is very aggressive for both the Moon and Mars. Thousands of tons of industrial equipment are delivered to the Moon, driving lunar cargo trips up to five per year. Lunar oxygen is placed in production as early as possible. One crew trip per year leads to a population of 30 because crew stay times on the Moon increase to several years.

Initial Mars missions use a cryogenic/all-propulsive system because the aggressive nature of the scenario merited an initial Mars mission as early as possible, and the reference nuclear electric propulsion system cannot be ready in time. The NEP missions are operated in a crew rotation/resupply mode, opposition profile, with each crew staying one synodic period (about 2.2 years). The reference scenario evolves to reusable MEVs based on Mars, fueled from Mars resources. Heavy cargo capability is provided, up to 250 t. per opportunity by 2020. The Mars population grows to 24, and by the end of the scenario can continue to grow by 24 or more per opportunity.

Industrialization and Settlement Program



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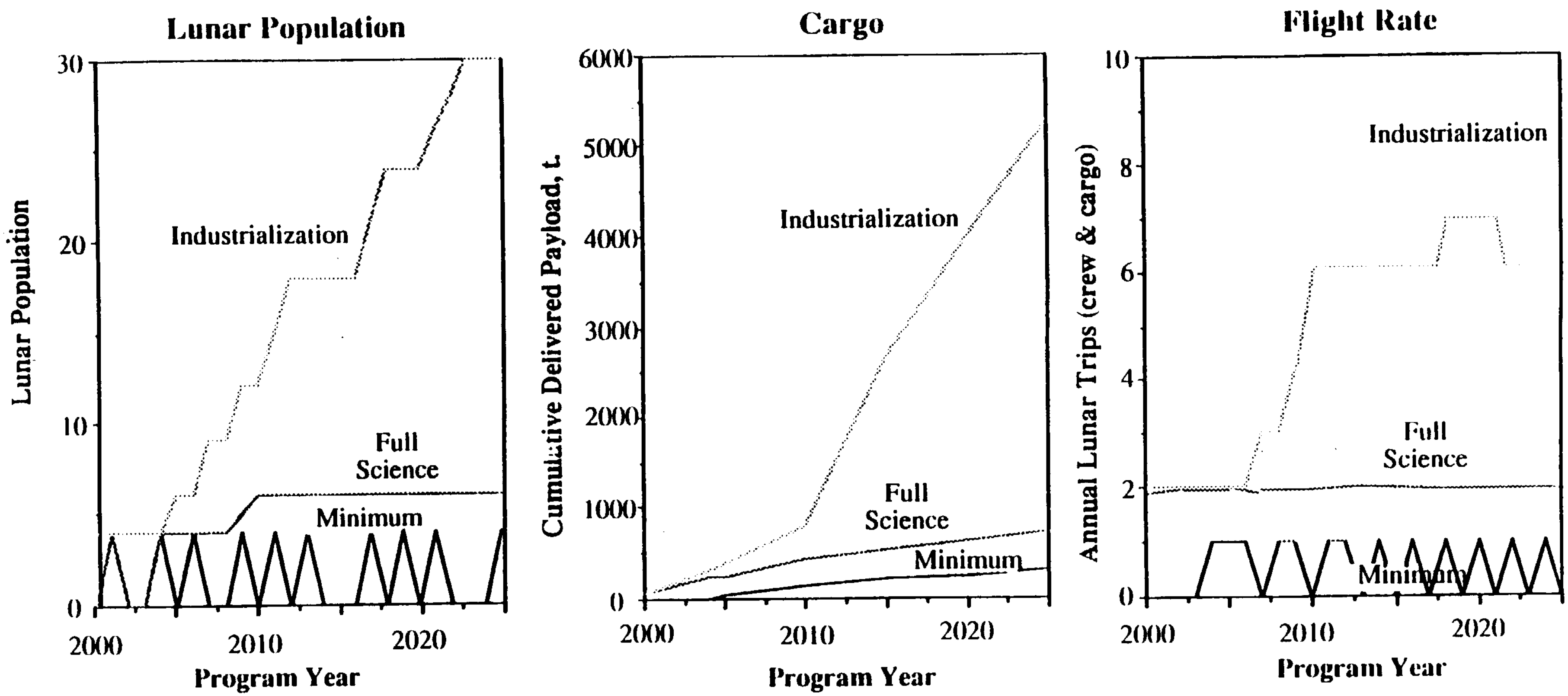
Lunar/Mars Program Comparisons

The next two charts compare the lunar and Mars program scenarios in terms of population, cumulative cargo delivered, and flight rate. The lunar population for the minimum scenario is four people for 30 to 40 days about every other year. The Mars population for the minimum scenario is 6 people on each of 3 conjunction missions, with 30 to 40 day surface stays. The full science menu scenario grows to year-long surface stays on conjunction missions. The lunar industrialization program goes to long stay times with indigenous food growth to build population. The Mars proto-settlement program obtains continuous presence by operating the NEP on an opposition-like profile in crew rotation/resupply mode. Later in this scenario, a second NEP is operated to provide two trips to Mars each opportunity.

- These scenarios were the "input" to the manifesting and life cycle cost analyses.

Lunar Program Comparison



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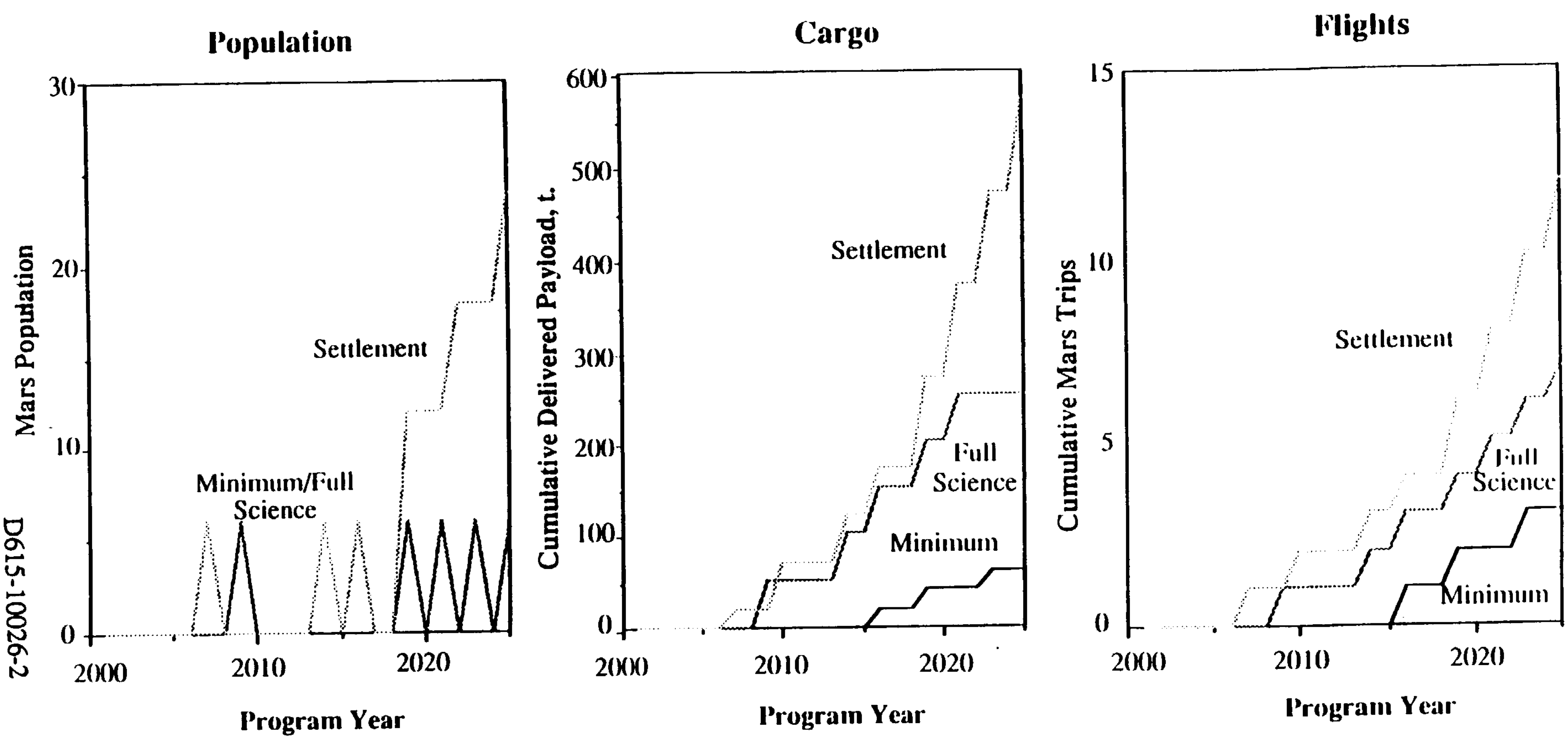
Mars Program Comparisons



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Architecture/Launch Vehicle/Node Trends

Issues

- Launch vehicle size, shroud size, and lift capacity.
- Node complexity and cost.
- On-orbit assembly complexity
- Number of launches per year
- Development cost
- Per-mission cost

Trends from Architecture Analyses

- Large launch vehicle (up to 300 t. lift) does not eliminate on-orbit assembly.
- Keys to on-orbit assembly are (1) design the vehicle to keep it simple; (2) design for automation and robotics; (3) reusable space vehicles to reduce the frequency of assembly operations.
- Advanced in-space transportation technology reduces launch requirements enough that a 100-t., 10-meter shroud launch vehicle is adequate.
- Ultra-large launch vehicle results in high early program costs and is much more costly than advanced in-space transportation technology.
- Evolution and design for evolutionary transitions are the keys to affordable, efficient programs with long-term growth.

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Available Options

The facing page is a typical listing of the element options making up a total transportation architecture for SEI missions. The options listed are all candidates for incorporation into architectures. Trade studies have not eliminated any of these options. (The list is representative and not necessarily complete.) The number of options on this chart for each row of options is indicated on the far right. In most cases, any option can be combined with any other set of options. Thus, the total possible combinations number in the millions. It is clear that available future effort can not hope to examine all combinations. This drives us to a strategy for architecture sensitivities analysis, to develop key trends and conclusions from relatively few architecture combinations.

Available Options

								No. of options
ETO	100 t.	140 t.	200+ t.	Add prop tanker				3 x 2
Node	SSF	Separate	SSF + separate	Self-assy.	Wet tanks	Refuel vehicles	Propellant depot	4 x 3
Lunar mode	Direct	Direct/ lunar ox.	LOR	LOR/ lunar ox.	L2/lunar oxygen			5
LTV	Cryo all-prop	Cryo aerobrake	NTR	NEP/SEP cargo	Fully reusable	Partially reusable	Expend-able	4 x 3
LEV	Cryo	Storable	Combined with LTV	Fully reusable	Partially reusable	Expendable		3 x 3
Mars mode	2.7 year	1.5 year						2
MTV	Cryo all-prop	Cryo aerobrake	NTR	NEP	SEP	Cycler		6
Mars node	LEO	L2						2
MEV	Cryo	Storable	Combined with LEV					3

Total possible combinations 2,799,360

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Top-Level Trade Table

The facing page considers mission profile, basing at Mars, and propulsion. Four important issues are central to mission profile selection: crew radiation exposure, crew time spent in zero g, the component of mission risk that increases with mission duration, and the added cost of shortening trip time. At one extreme is the notion, frequently expressed, that a Mars round-trip mission should be completed in a year or less. This is possible with certain advanced propulsion technologies, but at considerably higher cost than for longer trips, as described later in this section of the briefing. At the other extreme, trip time is seen as much less important than minimum mass and cost; conjunction profiles should be used. Crew time in zero g can be minimized by arrtificial-g spacecraft design. Increase in risk with duration is difficult to quantify. The mission duration issue presently is concerned mainly with cosmic ray exposure.

Crew radiation exposure comes from solar proton events (flares) and galactic cosmic rays, and from manmade sources if nuclear propulsion or power are used. Unshielded energy deposition from GCRs varies from 50 to 100 milligray (5 to 10 rad) per year. The low end of the unshielded range does not constrain Mars mission architectures, but the high end exceeds the present NCRP astronaut radiation guideline of 500 millisieverts/yr (this guideline is for space shuttle and space station missions; no guidelines have been given for Mars missions). It is possible that guidelines will be reduced in the future.

Five profile options are presented. Conjucntion fast transfer implies transfers much less than one year. Opposition/swingby trajectories vary from about 440 to about 550 days. Opposition/fast profiles imply 450 days or less, without swingby. The split sprint is a variation on the fast opposition profile in which the MEV and propellant for the return from Mars are sent in advance on a low-energy profile.

If galactic cosmic ray exposure must be controlled, we must either provide shielding on the transfer vehicle crew habitat or reduce exposure times. Shielding the transfer vehicle habitat dramatically increases its mass, requiring high performance propulsion such as nuclear, or favoring a cycler concept where massive habitats are emplaced on a suitable repeating trajectory and left there. To reduce exposure time, the applicable profiles are: (a) conjunction missions with fast transfers, i.e. less than 180 days, (b) fast opposition profiles, e.g. less than 1-year round trip, and (c) Mars surface rendezvous (Mars direct). The cycler/semi-cycler architectures offer shielding on the Earth-Mars leg, typically 5 months, and provides a 5-6 month conjunction transfer on the return trip. During the long stay at Mars, the crew must be on the surface most of the time unless a shielded Mars orbit habitat is also provided.

Fast-transfer conjunction missions may require orbit basing. A surface rendezvous mission may not be able to achieve the fast return transfer direct from Mars' surface with reasonable vehicle mass, because of the higher delta V required and because the payload launched from Mars' surface is the entire Earth return habitat rather than a lightweight, short-duration crew cab. Available propulsion options become very limited for fast missions. At one year, the only sensible options are NTR splits, where return propellant is prepositioned at Mars on a low-energy profile, or the use of a nuclear gas-core rocket. Below one year, the gas-core rocket quickly becomes the only option.

Top-Level Trade Table

Mission Profile	Propulsion				Basing	
	Cryo/ All-Prop	Cryo/ Aerobrake	NTR	NEP/ SEP	Orbit	Surface
Conjunction Minimum Energy	√	No advantage over propulsive capture	√	√	√	Later
Conjunction Fast Transfer	Excessive IMLEO	√	√	√	No. Reason for fast transfer is less GCR dose	√
Opposition/ Swingby	Same	√	√	Note 1	√	As a resupply mode
Opposition/ Fast	Same	Excessive IMLEO	√	Not able to make fast trips	√	Same
Opposition/ Split Sprint	Same	Same	√	Cargo only	√	Same

Note 1: NEP flies an opposition/swingby-like-profile but does not benefit from Venus swingby.

Architecture Results for Three Activity Levels

The top-level architecture selection results for the three activity levels are shown on the facing page. For the minimum program, a cryogenic expendable tandem-staged direct mode is the clear economic winner. Its lower development expense causes the operational cost savings for a reusable LOR system to have little payoff. At the median activity level, the reusable system gives about a 5% return on investment (ROI). Our baseline program included lunar oxygen at the median level, but the ROI is estimated only about 3%. At the high lunar activity level, reusable systems and lunar oxygen both have strong payoff, e.g. the lunar oxygen ROI is about 10%

The minimum Mars program is most economic with cryogenic all- propulsive expendable vehicles on conjunction profiles. The NTR has an ROI less than 2% at this level. If natural environment radiation concerns lead to a conjunction fast transfer or opposition profile, the NTR is the preferred solution with cryogenic/aerobraking as a backup. At the median level, the NTR has a 16% ROI versus cryo all-propulsive. Here also, aerobraking is a backup and SEP comes into the picture as a "dark horse", with about 10% ROI if array costs can be reduced to $100/watt, a tenfold reduction from present costs. At $500/watt, the SEP has a negative 10% ROI, showing the great leverage of array cost. At the high level, electric propulsion is indicated as important, but development costs are a problem unless low-cost SEP arrays can be produced. If electric propulsion costs are too high for a settlement-scale Mars program, the NTR/dash and Mars direct modes are viable options.

Architecture Results for Three Activity Levels

Minimum

Lunar:

Expendable

Mars:

- Cryogenic all-propulsive
- Unless radiation environment requires reduced trip times; then nuclear rocket or cryo aerobrake conjunction fast transfer

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Median (full science)

Lunar:

Start expendable, possible growth to LOR reusable, aerobraking

Mars:

- Nuclear rocket, conjunction, multiple landers
- Opposition or conjunction fast transfer options
- Cryo/aerobraking backup
- SEP "dark horse"

Industrialization /settlement

Lunar:

LOR crew and tandem direct cargo, reusable, with lunar oxygen

Mars:

- Early cryo/all-propulsive option
- Electric propulsion for sustained growth (probably SEP)
- Nuclear rocket/dash or Mars direct/Mars propellant, options for crew rotation and resupply.

Seven Architecture Recommendations

The next seven pages contain our main architecture recommendations with data illustrating key points.

Lunar Architecture

2

- <u>Begin the lunar program with a tandem-direct expendable system.</u>
  - System can be designed to eliminate on-orbit assembly; one docking or berthing required.
  - The number of development projects is minimized. Offers reasonable expectation of return to the Moon by 2004 under likely funding constraints.
  - Flight mechanics constraints for LOR operations are avoided.
  - Tandem-direct LTV is a starting point for evolution to all other identified lunar architectures.
  - Lunar aerobrake can be tested on the unmanned booster stage without risk to the crew. Stage is otherwise expended.

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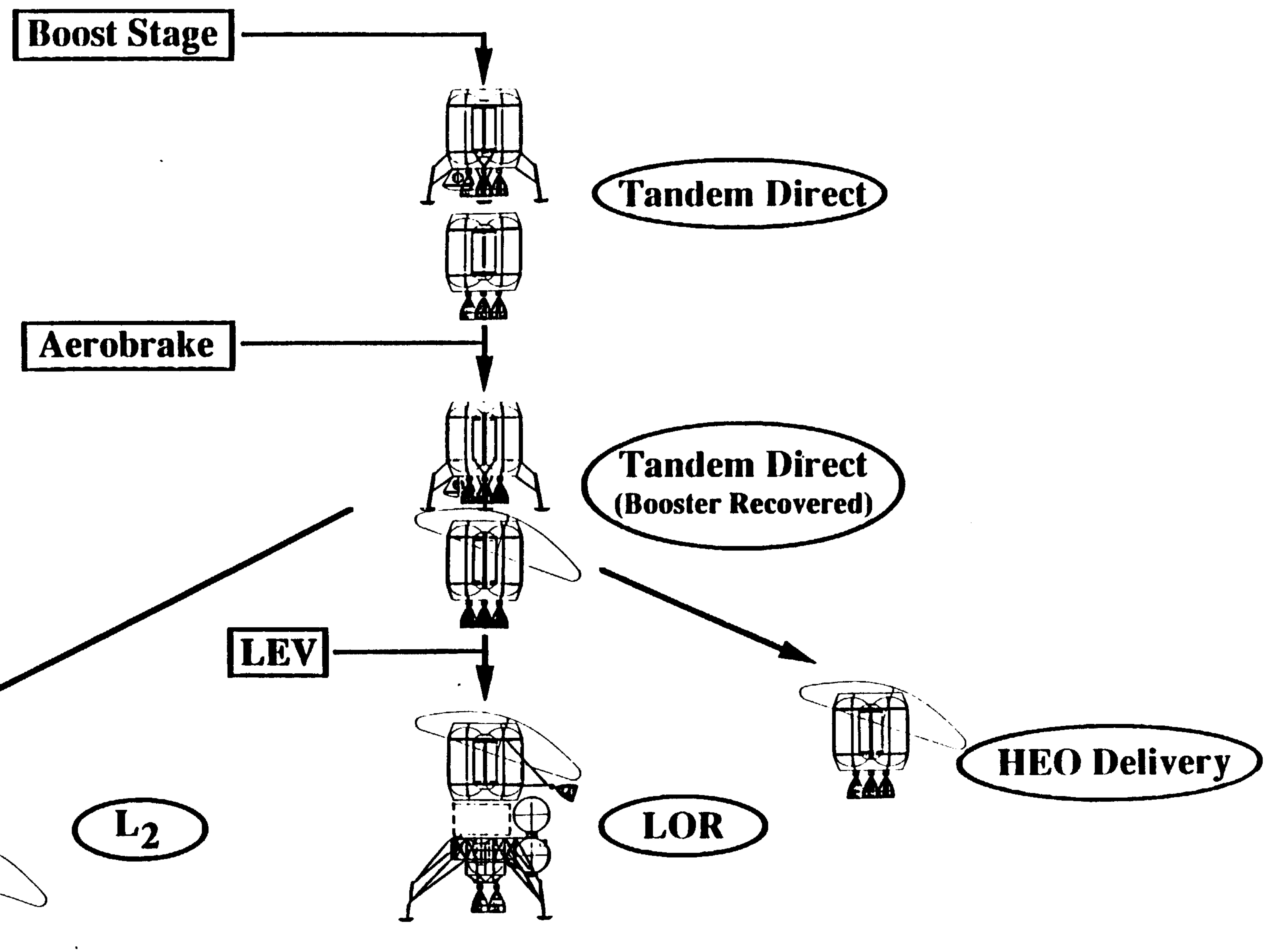
Lunar Transportation Family
Evolution



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Lunar Cryogenic Propulsion

- <u>Invest in cryogenic storage and management technology.</u>
  - Without advanced development of a low-boiloff flight-weight cryogenic insulation system, the lunar program may be forced to a storable propulsion system for lunar vicinity operations. Cost impact is billions of dollars.

- <u>Invest in a 30K-class advanced expander cryogenic engine with 10:1 or better throttling capability.</u>
  - An advanced expander engine offers about 20 seconds' Isp gain over a modified RL-10; can demonstrate advanced health monitoring and maintainability features essential for Mars missions.

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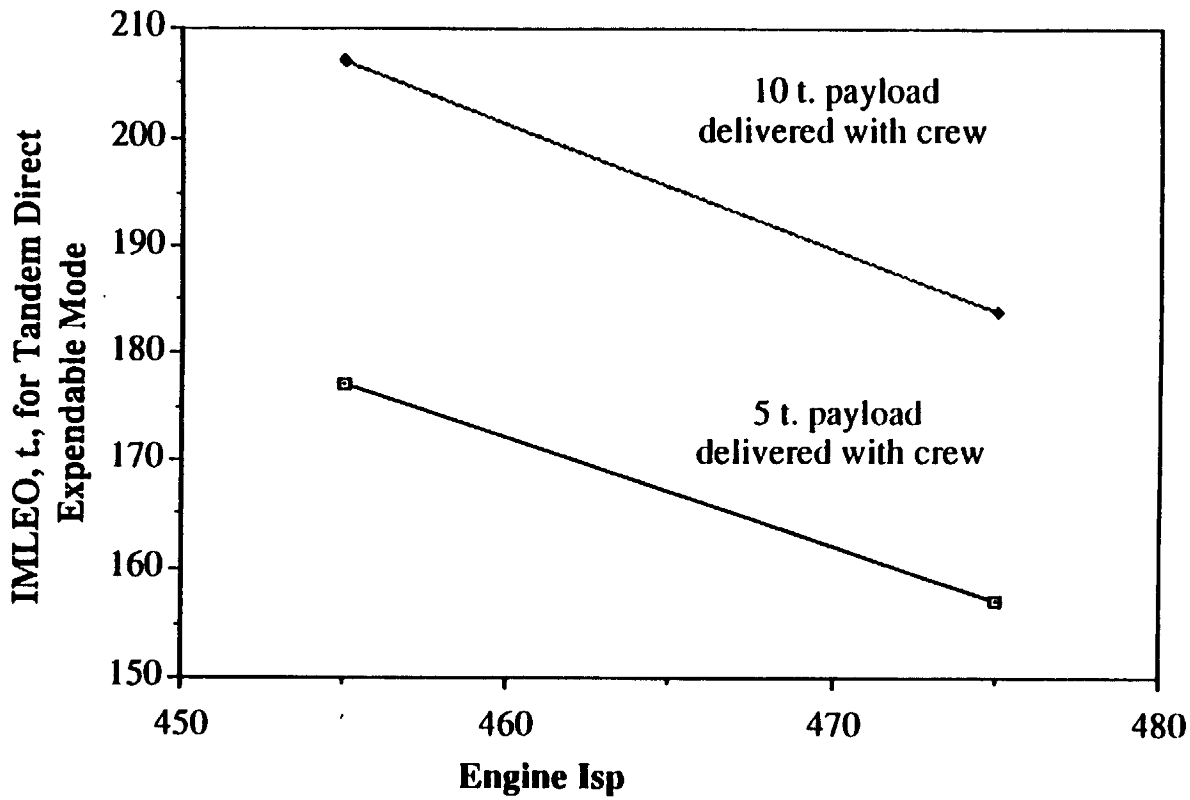
Early Lunar Mode Sensitivity to Isp



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Mars Baseline Architecture

- **Baseline nuclear thermal rocket propulsion for Mars.**
  - Nuclear thermal rocket indicated as very economic and flexible over wide range of program activity levels.
  - Nuclear rocket vehicle mass is sensitive to specific impulse. Isp gain for carbide fuels is well worth the technology investment.
  - Development and qualification testing requires proven test facility technology that contains hydrogen effluent and scrubs radioactivity.
  - Nuclear rocket performance permits modest lunar program and significant Mars exploration with about six launches per year of 100-tonne class HLLV.
  - Nuclear rocket baseline offers reasonable expectation of initial Mars mission by 2010 under likely funding constraints.

- **Recommended technology advancement program:**
  - High-performance fuels
  - Full-containment ground test facilities.

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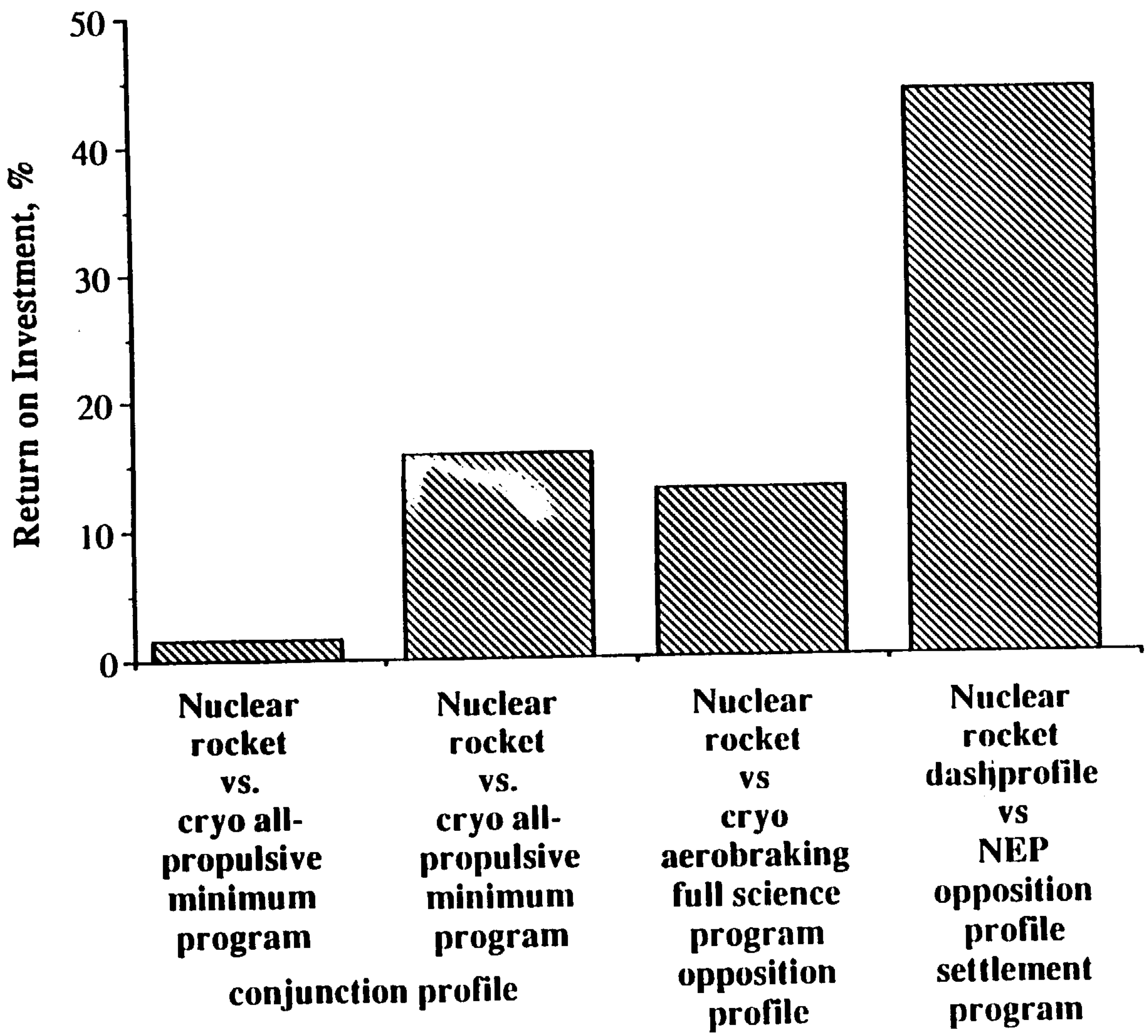
Nuclear Rocket ROI Trades



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Aerobraking Technology

- <u>Accelerate aerobraking technology for Mars aerocapture as backup to nuclear rocket.</u>
  - Target decision between the two in the 1996-2000 time frame.
  - NTR performance and cost uncertainties, especially test facilities and testing, merit backup.
  - Aerobraking needed for Mars landing. Technology challenges less daunting than aerocapture, but merit technology program.
  - Aerobraking technology keeps other options open.
    - Conjunction fast transfer
    - Mars direct
    - Cycler orbits
    - NTR-dash profile
  - Aerobraking is economic for lunar transportation at >= two flights/year.

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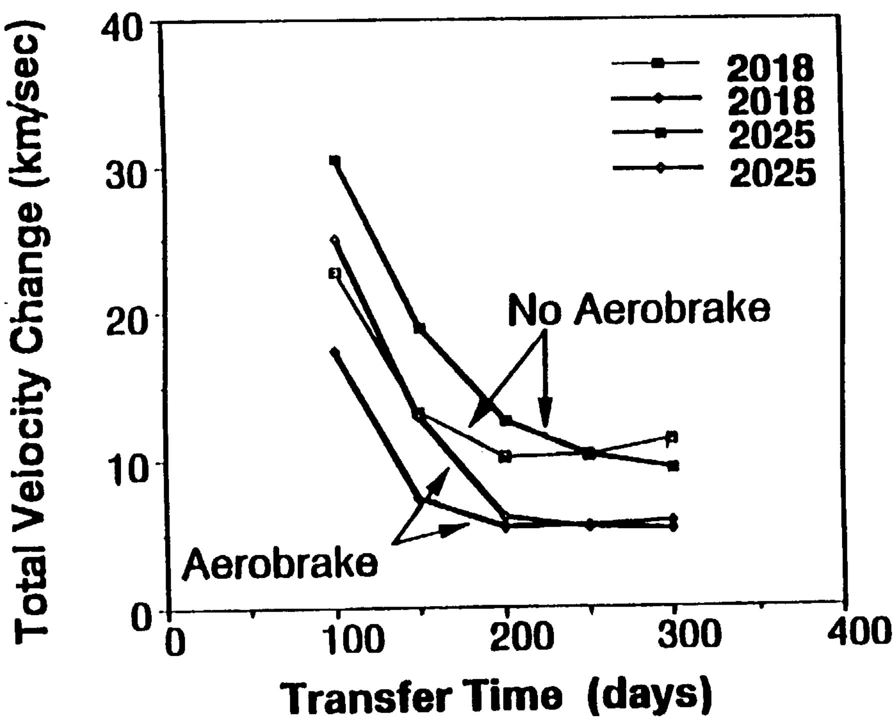
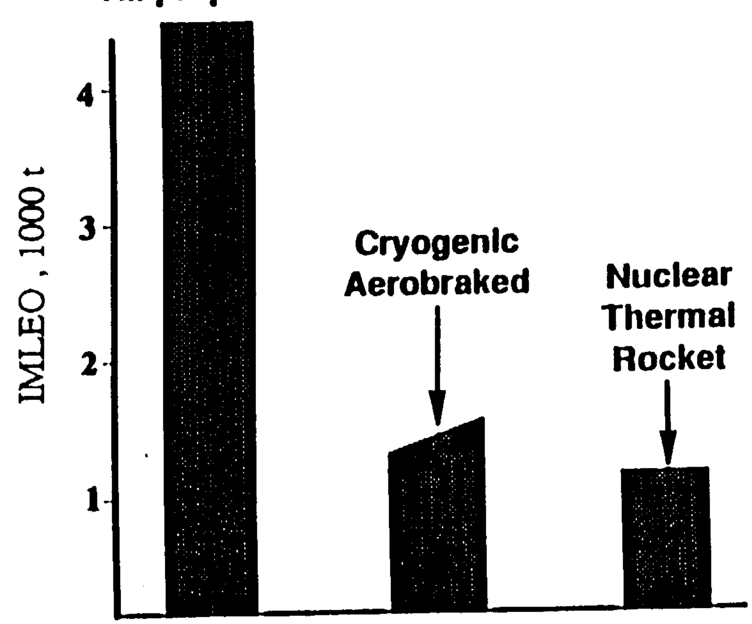
Mars Aerocapture Benefit for Conjunction Fast Transfer





Conjunction Fast Transfer

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Program Implementation Architectures Relation to Aerobraking

The facing page indicates uses of aerobraking for the various architectures. As noted, some form of aerobraking occurs in all of the architectures, in particular for Mars landing and for Earth capture on return from lunar missions. In addition, some of the architectures include an Earth crew capture vehicle (ECCV) for direct return of the crew to Earth in cases where, for example, an NEP or SEP vehicle must spiral back down to LEO or in the case of an NTR where the vehicle captures into a highly elliptic orbit.

Program Implementation Architectures

Architecture	Features	Aerobraking Functio Mars cap	Mars land	Earth cap/ lunar	Earth cap/ Mars	Earth entry*
Cryogenic/aerobraking	Cryogenic chemical propulsion and aerobraking at Mars and Earth. LEO-based operations.	x	x	x	x	x
NEP	Nuclear-electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
SEP	Solar electric propulsion for Mars transfer; optionally for lunar cargo.		x	x		x
NTR (nuclear rocket)	Nuclear rocket propulsion for Lunar and Mars transfer.		x	x		x
L2 Based cryogenic/ aerobraking	L2-based operations; optional use of lunar oxygen.	**	x	x	x	x
Direct cryogenic/ aerobraking	Combined MTV/MEV refuels at Mars and LEO. "Fast" conjunction profiles.	x	x	x	x	
Cycler orbits	Cycler orbit stations a la 1986 Space Commission report	***	x	x	x	x

Notes: * optional/emergency mode **opposition class only *** MEV-class crew taxi (not a large MTV)

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Aerobraking Flight Test Bed

- <u>Perform aerobrake tests on the LTV booster, to put the technology on the shelf for Mars.</u>

  - If the lunar program grows to high activity levels, lunar aerobrake is economically justified.
  - A space-assembled aerobrake is needed for Mars landing.
  - Aerocapture technology is needed as backup to Mars NTR.

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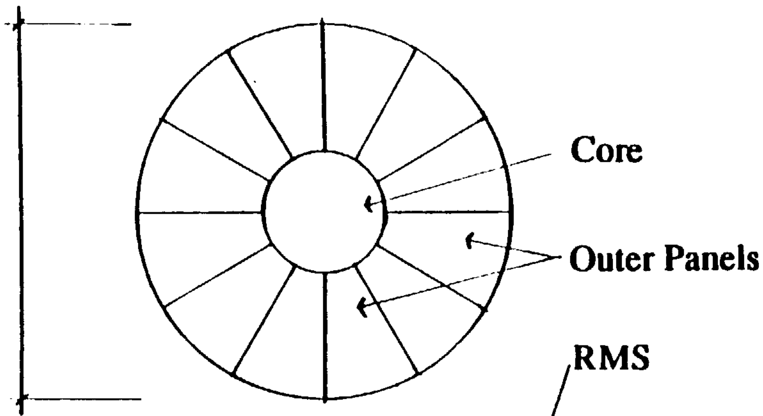
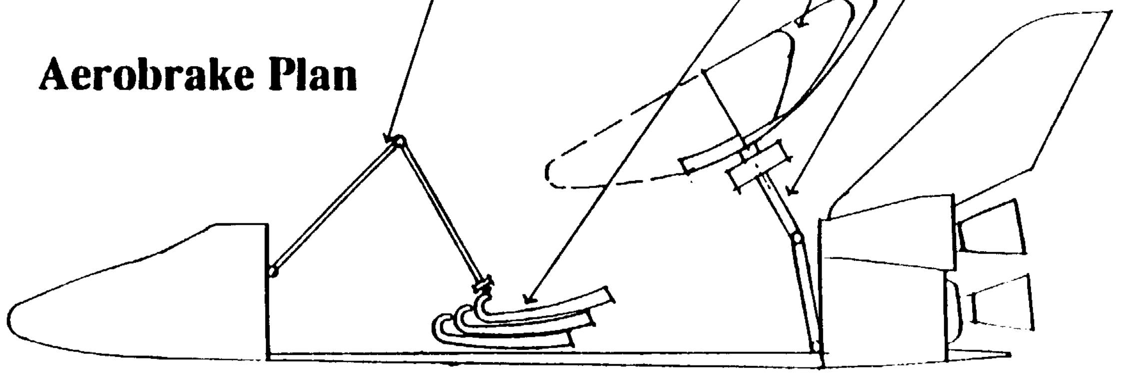
Aerobrake Assembly Test in LEO



Aerobrake Plan



Cross Section

Assembly arm rotates brake as outer panels are installed for easy RMS reach and crew visual contact during operations

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SEP as "Dark Horse"

- <u>Designate solar-electric propulsion (SEP) as a "dark horse" for Mars transportation.</u>

  - Technology advancement issues:

    - Light weight, high performance, radiation resistant arrays.
    - Automated production technology, $100/watt
    - Robotics technology for constructing SEP and deploying arrays
    - Long-life, high power density, efficient electric thrusters

  - If safety precludes operation of nuclear propulsion in low Earth orbit, SEP is the only option more economic than cryo-genic/aerobraking.

  - If low-cost array target achieved, SEP is more economic than NEP.

  - SEP is the most likely architecture for eventual private sector use for Mars settlement.

  - SEP technology has derivative benefits, e.g. power beaming to planet surfaces.

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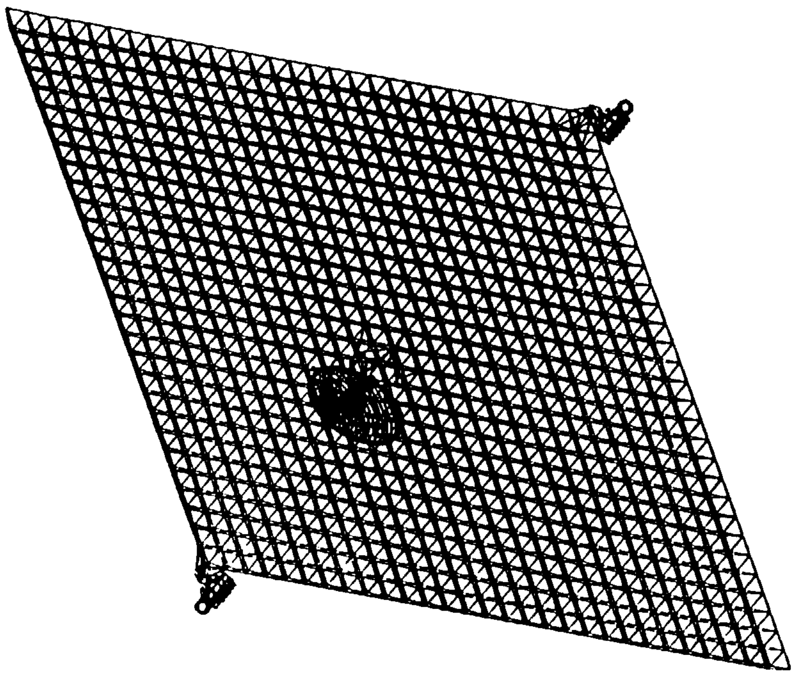
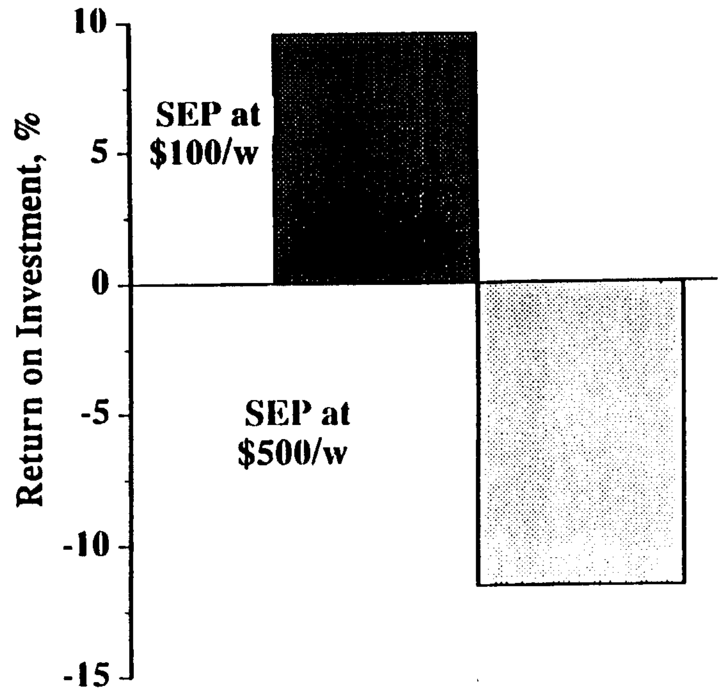
Return on Investment Comparison
SEP versus NTR; Full Science Program





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Nuclear Space Power

- <u>Continue the nuclear space power program towards near-term systems applicable to planet surface power.</u>
  - DDT&E and production cost estimates from this study eliminate nuclear electric propulsion (NEP) as a top contender, but are very preliminary.
  - As NEP systems are better understood, estimates may come down.
  - To keep NEP option open:
    - Further studies to better understand the cost of nuclear power systems suitable for electric propulsion.
    - Modest funding of high-leverage high-performance power conversion technology.

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Mission Risk Comparison

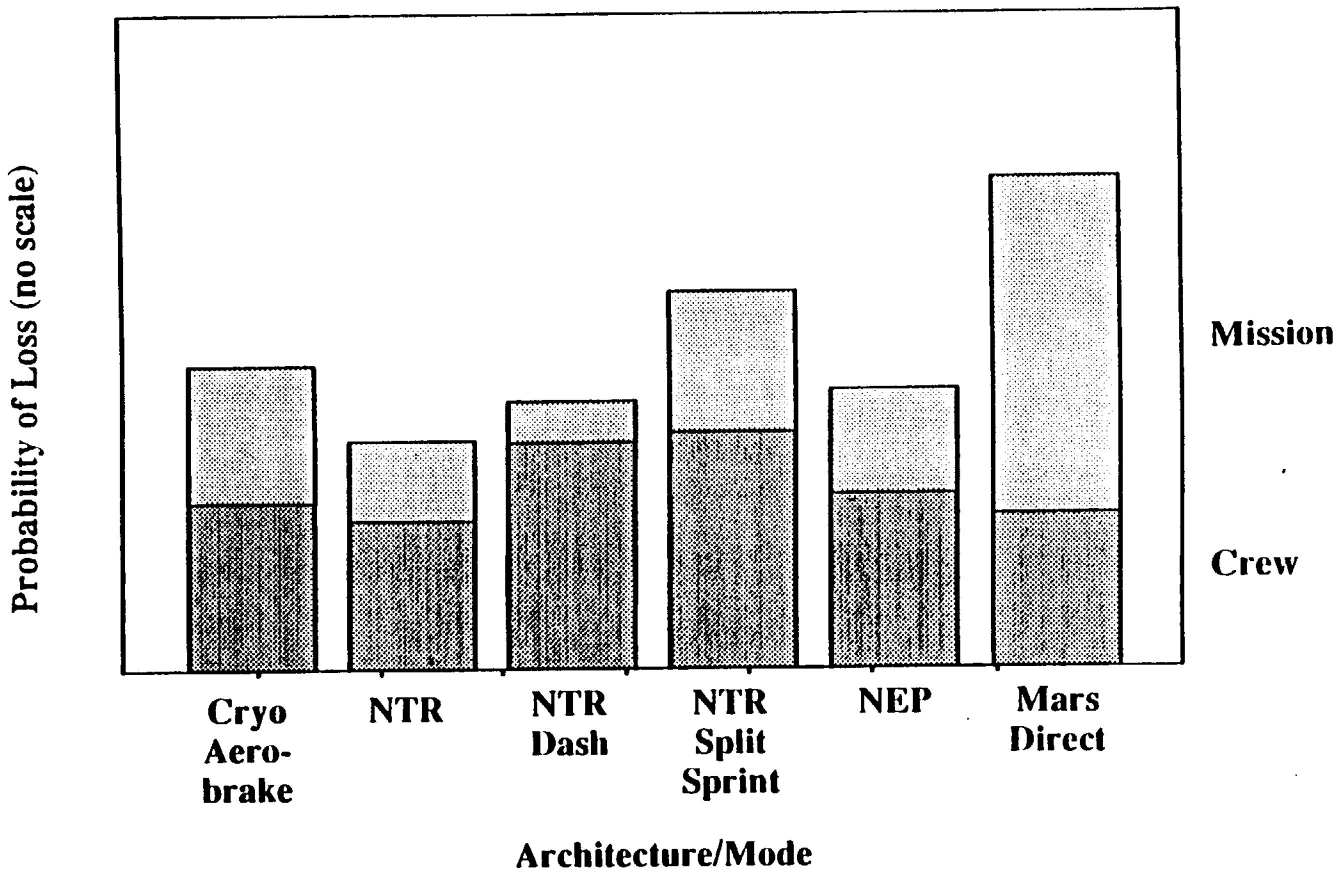
Mission risks were compared in a semi-quantitative way. The methodology is rigorous and quantitative, but reliability and safety estimates for SEI hardware and maneuvers are no more than ballpark guesses today. We made representative estimates with an attempt to be consistent, i.e. the same type of maneuver was given the same number for all cases. Plausible differences were used, e.g. aerocapture was judged higher risk than propulsive capture. Abort modes were included where available.

The facing page shows comparative risks for crew loss and mission loss for several architectures and modes.

NTR shows the least risk because of the propulsive capture advantage, and because a free return abort was assumed, as it was for the cryo/aerobrake. The NTR/dash mode does not permit free return abort or descent abort at Mars, so some mission loss risk turns into crew loss risk. As Mars transportation matures and a safe refuge on the surface of Mars is available, the NTR/dash mode is deemed acceptable. The NTR split sprint mode also exhibits higher risk because of lack of abort modes, e.g. no free return. NEP is shown comparable to, but slightly riskier than NTR. The NEP case is sensitive to the lifetime dependability of the propulsion system; this figure is much more uncertain than NTR reliability. Mars direct has a higher mission loss risk because of its complex automated operations, but the crew loss risk is comparable to the others. The perception of crew loss risk for Mars direct is probably higher than the real risk.

(1-2)

Mission Risk Comparison



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Man Rating Requirements

The facing page describes our recommended approach to man-rating and lists the systems/subsystems for which we believe man-rating is required.

Man-Rating Requirements

Approach

- Ground-based testing wherever possible.
- Use flight program activities to bootstrap, e.g. lunar aerobrake program builds confidence in Mars aerobrakes.
- Flight demonstration of critical functions, e.g. Mars cargo landing, before critical manned use.
- Life demo for long-duration systems before critical manned use, e.g. ECLSS on SSF or lunar surface before manned Mars mission.

Subjects

- Aerobrakes
- Cryogenic rocket engines
- Nuclear rocket engines
- Cryogenic propellant systems
- Attitude control propulsion systems
- Nuclear & solar electric propulsion systems
- ECLSS/TCS
- Crew modules/hab systems
- Vehicle power
- Avionics & Communications systems
- Surface transportation systems

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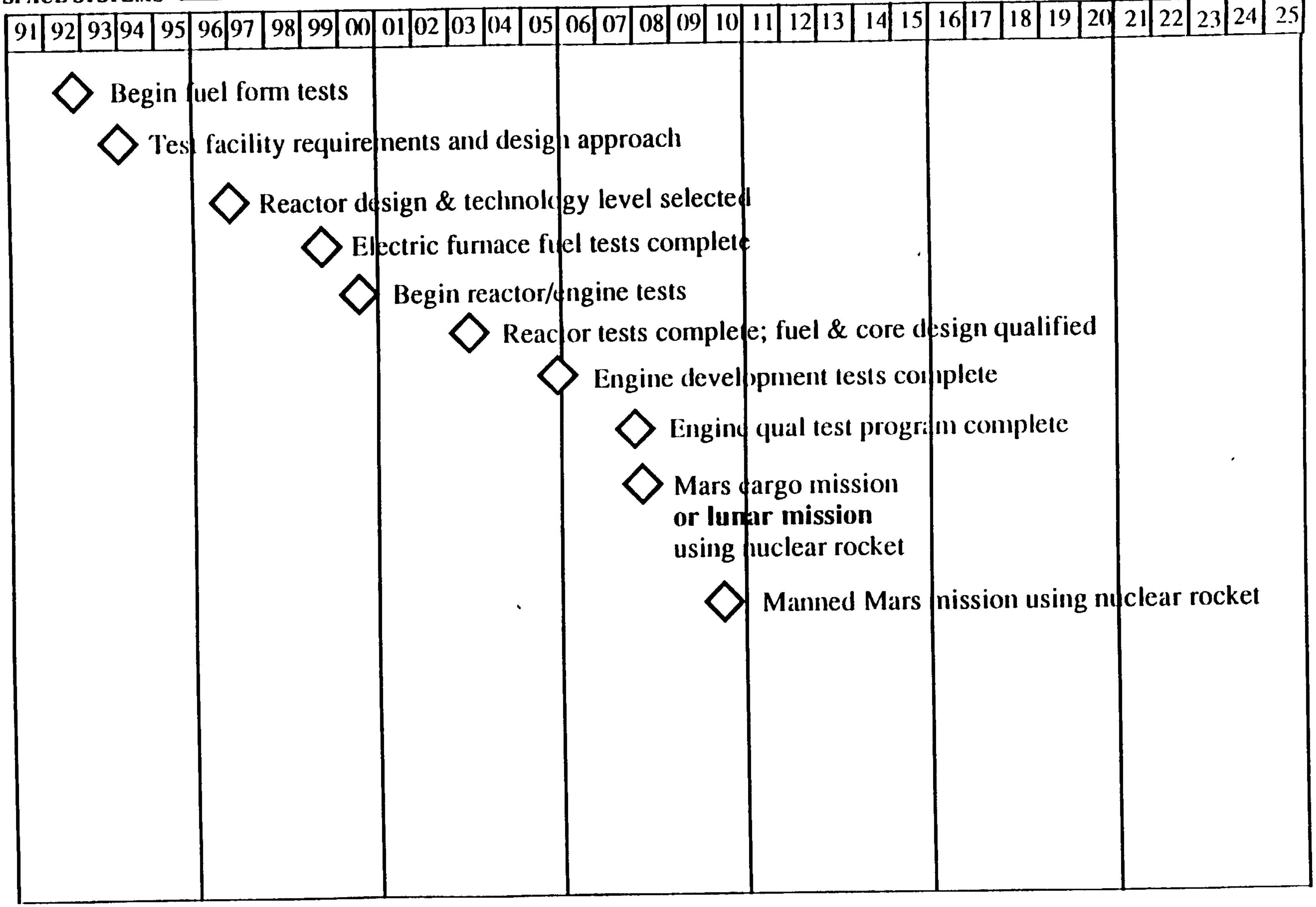
Nuclear Rocket Man-Rating Approach

A sequence of major tests and demonstrations to achieve nuclear rocket man-rating is shown. Note that two flight demonstration options exist. A decision of which to use depends on whether cargo delivery to Mars is needed before the first manned mission, as would be the case if a conjunction fast transfer and long surface stay is required on the first mission to reduce galactic cosmic ray exposure to the crew.

Nuclear Rocket Man-Rating Approach



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Technology Advancement and Advanced Development

The next three charts present our current recommendations for technology advancement and advanced development, with schedules and funding estimates. The funding level averages about $300 million per year. If we consider the median (full science) program as representative, the technology/advanced development program is about 0.2% of the life cycle cost of the program to 2025, a very modest investment.

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Technology Development Schedules
- Overview -

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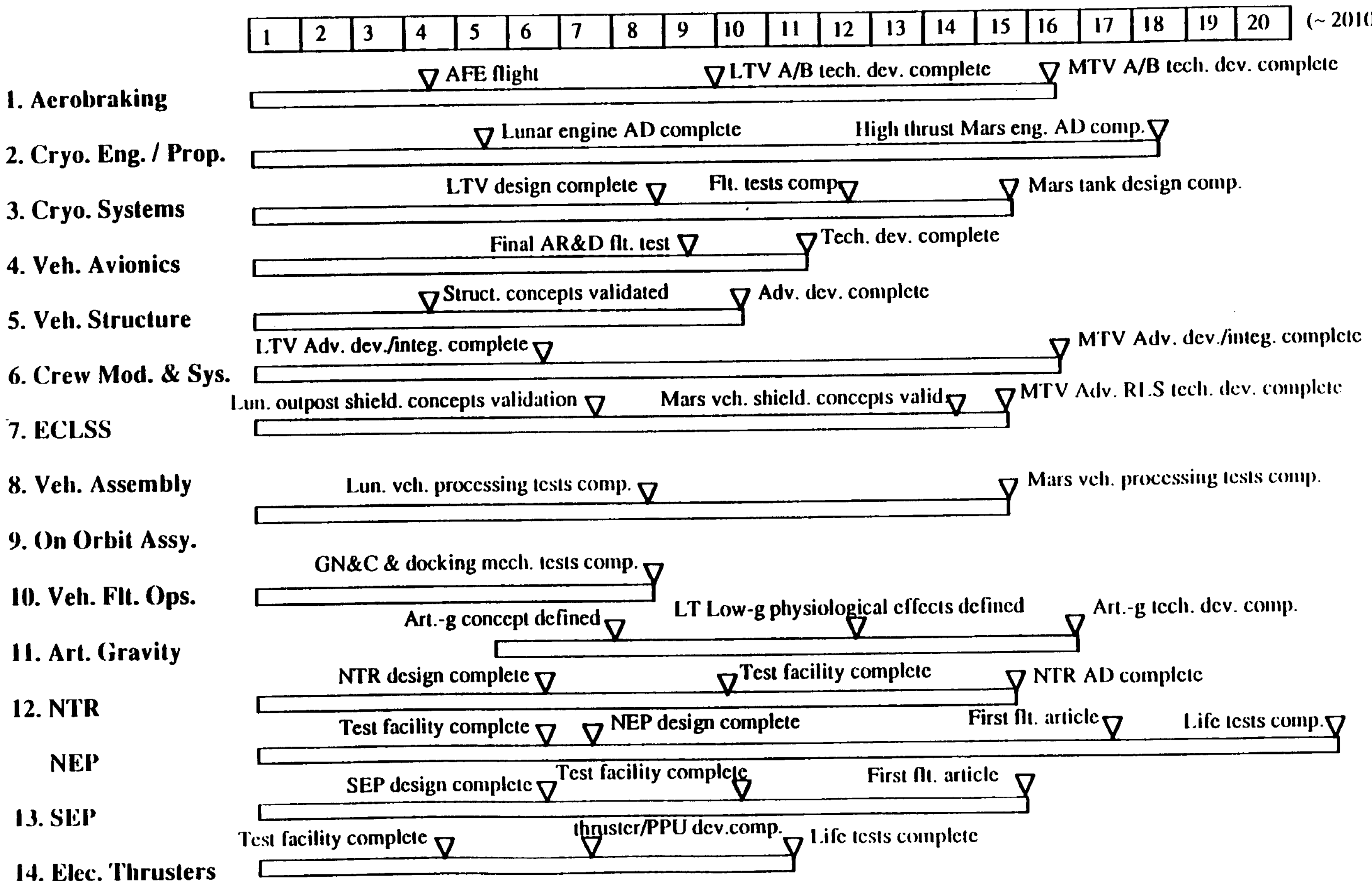

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Technology / Advanced Development Funding Estimates

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
1 - Aerobraking* -Technol.	1	6	10	5	5	8	10	10	8	5		68 M
- Adv. Dev.	0	0	30	30	55	20	30	65	65	40	30	400 M
2 - Cryogenic Engines / Prop.	0	30	30	30	20							110 M
- Adv. Dev.	0	0	99	71	65	50	65	65	50			465 M
3 - Cryogenic Systems - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	10	10	20	50	50	50	110				300 M
4 - Vehicle Avionics/Software	2	5	5	5								17 M
- Adv. Dev.	0	0	0	25	45	40	40	40	40	40		270 M
5 - Vehicle Structures - Tech.	3	7	5	5	7	7	5					39 M
- Adv. Dev.	0	0	15	17	11	10	15	15	15	10		108 M
6 - Crew Modules & Systems	0	0	3	3	3	5	5	5	3			27 M
- Adv. Dev.	0	0	15	20	10	10	15	20	20	10		120 M
7 - Environ. Ctrl. & Life Supp.	0	0	3	5	5	10	10	5	5			43 M
- Adv. Dev.	0	0	6	10	10	15	30	30	30	20		151 M

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Technology / Advanced Development Funding Estimates

Technology Category	1	2	3	4	5	6	7	8	9	10	11	Total
8 - Vehicle Assembly - Tech.	5	5	5	5								20 M
- Adv. Dev.	0	5	40	40	40	40	40	40	10			255 M
9 - Orbit Launch & Checkout	5	5	5	5								20 M
- Adv. Dev.	0	4	15	16	5	10	10	10	10	5		85 M
10 - Vehicle Flight Operations - Adv. Dev.	0	0	9	15	10	15	15	15	10	5		94 M
11 - Artificial Gravity - Tech.	0	0	0	2	5	10	10	10	10	3		50 M
- Adv. Dev.												
12 - Nuclear Propulsion												
NTP -	0	10	15	20	20	20						85 M
NEP -	0	15	20	30	30	30	20	20				165 M
13 - Solar Electric Ion Prop.	2	8	10	15	15	10						60 M
Array manufac. Tech. -	0	0	0	30	30	30						90 M
14 - Electric Thrusters	0	5	10	20	20	20	10					85 M
Tech. Development Total	**23**	**120**	**367**	**482**	**461**	**410**	**380**	**460**	**276**	**138**	**30**	**3147 M**

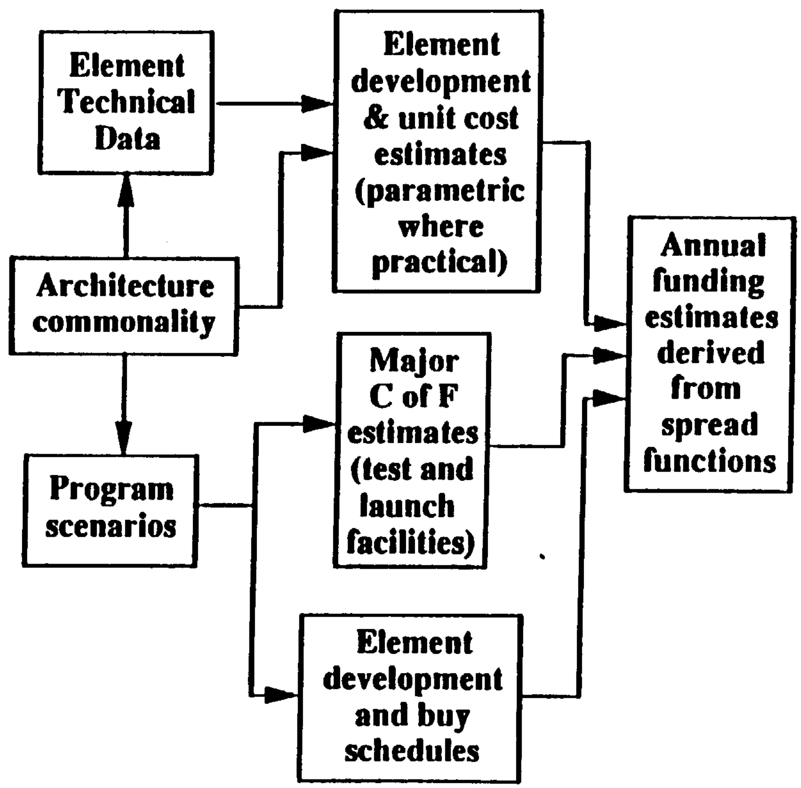
Life Cycle Cost Model Approach

Our basic cost model kernels are parametric cost models. We use the Boeing Parametric Cost Model and the RCA Price models to estimate development and unit cost. The determination of hardware to be costed comes from what architectural elements are needed and from element commonality of the architecture. Program schedules determine requirements and timing for major facilities and for the element development and buy schedules. All of these inputs are used to estimate annual funding for each component of the program, using cost spread functions. The costs are integrated into a spread sheet life cycle cost model to obtain annual funding for complete programs.

The ground rules used in this analysis are indicated on the chart.

The ground rule for use of closed ecological life support (CELSS) and lunar oxygen comes from economics trade studies conducted several years ago through last year.

Life Cycle Cost Model Approach



Ground Rules

- No precursor missions costed.
- NASA contingency not added
- Common element in new application gets 25% delta DDT&E cost.
- No production learning unless production rate > 1 per year.
- Production rates maintained minimum of 1 per 5 years to keep lines open.
- Mission definitions flexible to enable transportation systems to operate at high efficiency.
- All scenarios include closed ecological life support and ISRU for efficiency.

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Architectural Cost Drivers

Our investigations of architectures, while preliminary, indicate the importance of cost drivers, in the order listed on the chart. The number of development projects should be minimized through commonality and phased by evolution so that development costs are reduced and are spread over the life cycle of the program, rather than lumped early in the program.

Space hardware for SEI missions is expensive and should be reused if possible. As an example, our unit cost estimate for the Mars transfer crew module is more than a billion dollars. Reuse of this equipment motivates investment in the advanced transportation technology needed to make it reusable.

The third point is that Earth launch mass drives Earth launch cost. Even if Earth launch cost is reduced by ALS-class vehicles, the Earth launch cost is the largest single part of program cost.

The final point is that design and development of systems with mission and operation flexibility enhances commonality and minimizes the risk that changes in mission requirements force new developments or major changes.

Architecture Cost Drivers

- Number of development projects (minimize through commonality)
- System reuse (maximize)
- Earth launch mass (minimize)
- Mission and operational flexibility (maximize)

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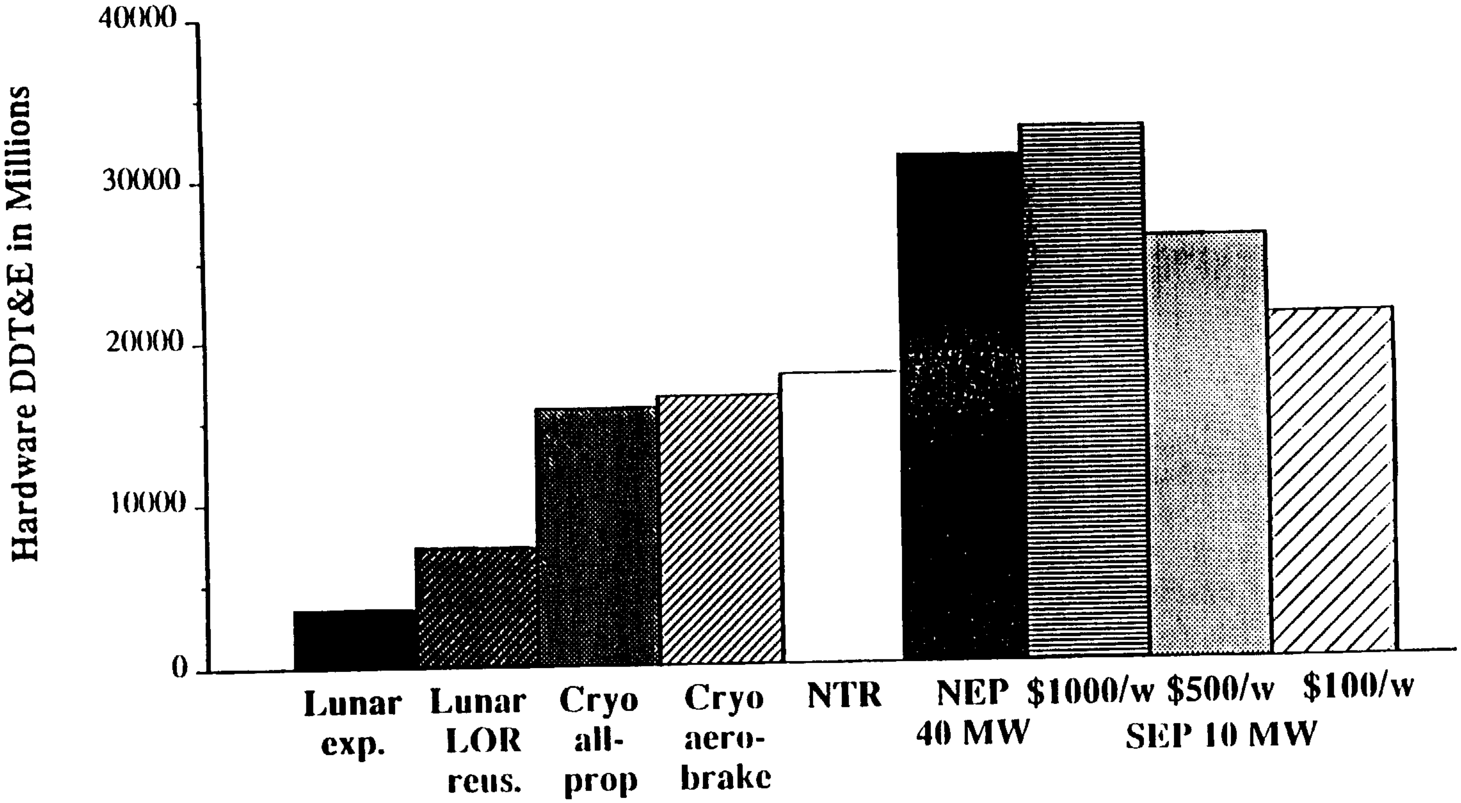
In-Space Transportation DDT&E Comparison



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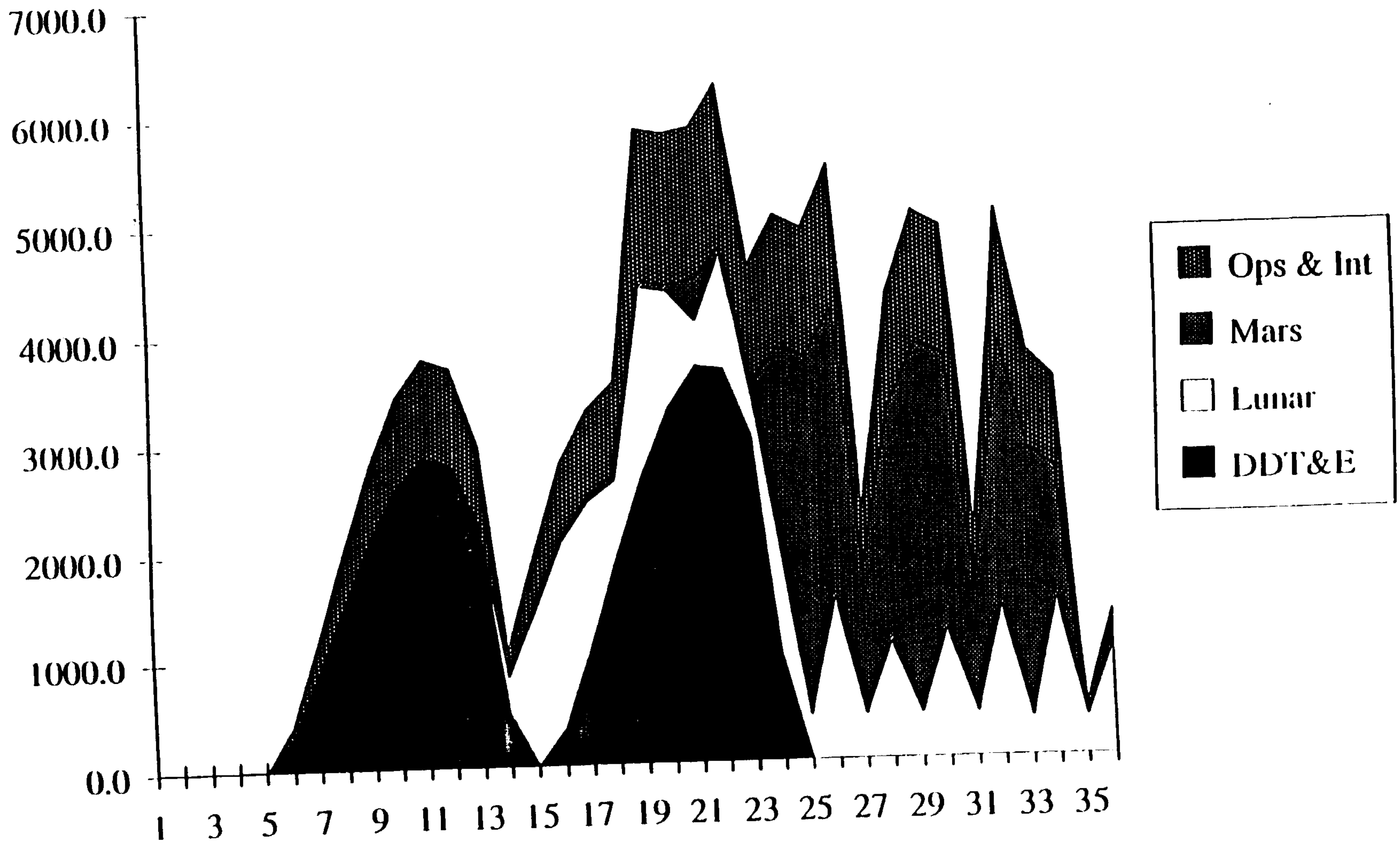
Minimum Program Life Cycle Cost Spread

The minimum program life cycle cost spread peaks between five and six billions per year. The deep valley between lunar and Mars peaks indicates that the Mars program should occur earlier in this program. The minimum program involves relatively modest investments in surface systems and falls well below the SEI funding wedge implied by the Augustine Committee recommendations.

Minimum (Baseline W/Ops & Int)



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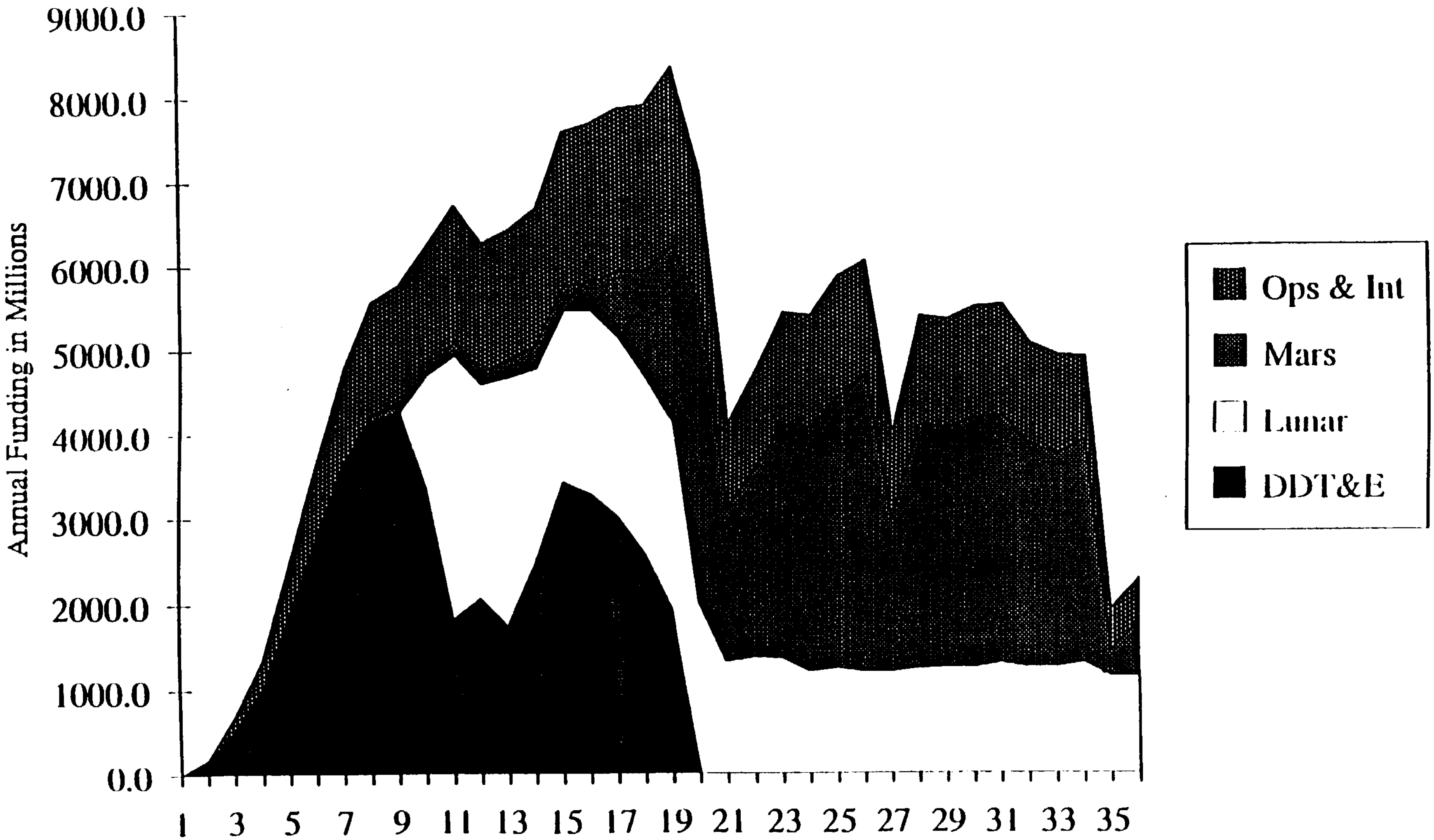
Median (Full Science) Program Life Cycle Cost Spread

The median life cycle cost spread peaks at about eight billions per year. With addition of likely surface systems costs, this program probably exceeds the Augustine guidelines during the peak years.

The median program exceeds by a factor of several the science and exploration potential of the minimum program. Lunar human presence grows from an occasional 45 days to permanent presence of six people, and Mars surface time grows from about four man- years to about 30. In other words, a roughly 50% increase in cost leads to about an order of magnitude increase in exploration and science potential.

Full Science (Baseline W/Ops Int)



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Median (Full Science) Program Life Cycle Cost Spread
Reduced Early Lunar Program

By deferring major lunar activities, the median program can be brought within the Augustine guidelines. Permanent human lunar presence is delayed until after the Mars DDT&E peak. The early lunar program is like the minimum scenario, i.e. man-tended astrophysics observatories.

Another way to level the funding profile for the median program is to defer Mars by a few years. The reference median program achieves a Mars landing in 2010 (2009 departure). Deferral to about 2016 would probably smooth out the funding profile much as did the reduction of the early lunar program.

Our view was that getting to Mars early was more important than an early buildup to permanent lunar presence. The partially deferred lunar program represented here still achieves astrophysical observatories early, but defers permanent human presence until after the major Mars mission DDT&E is complete.

ADVANCED CIVIL SPACE SYSTEMS

Full Science (Baseline W/Ops Int-Reduced Lunar)

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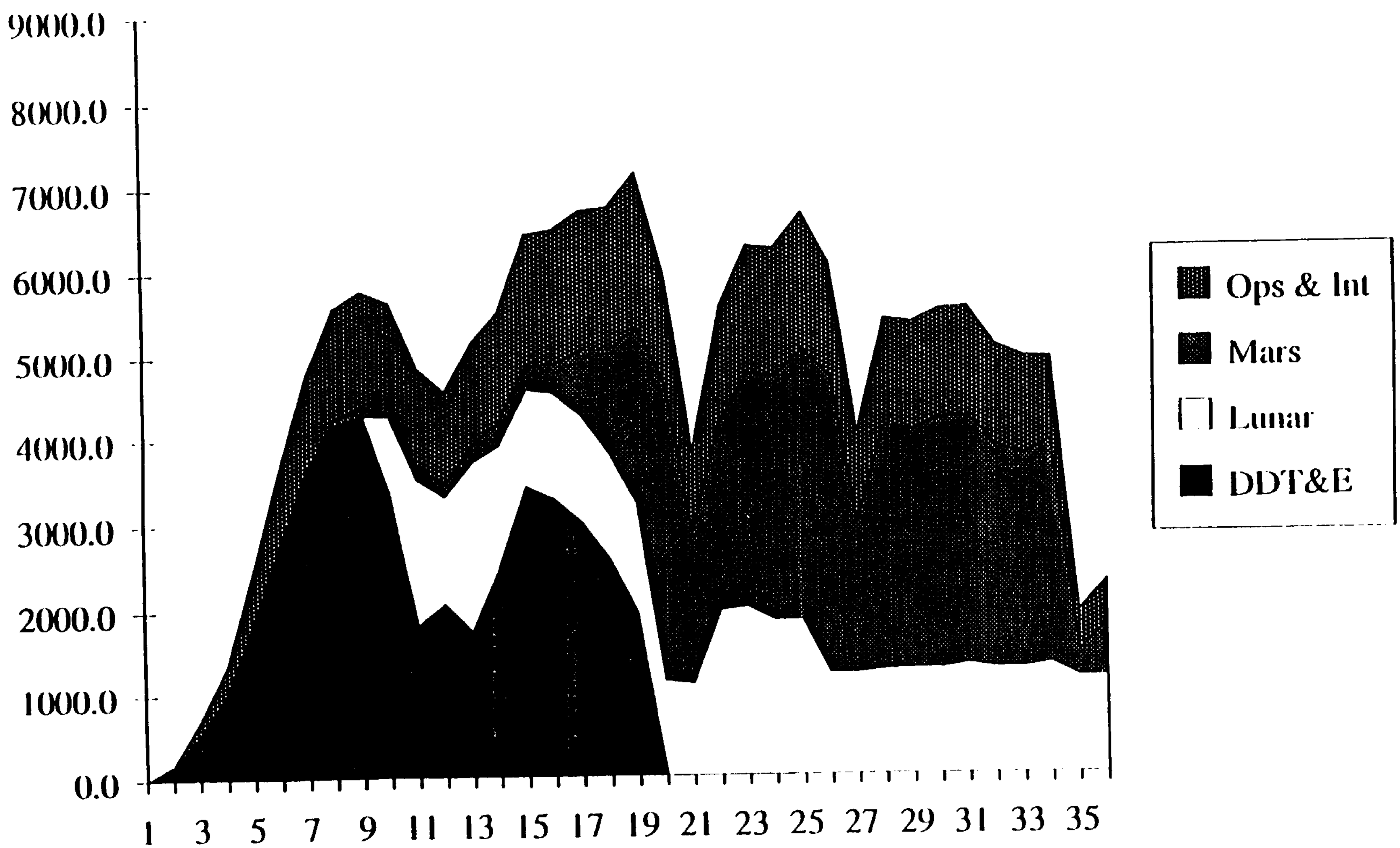

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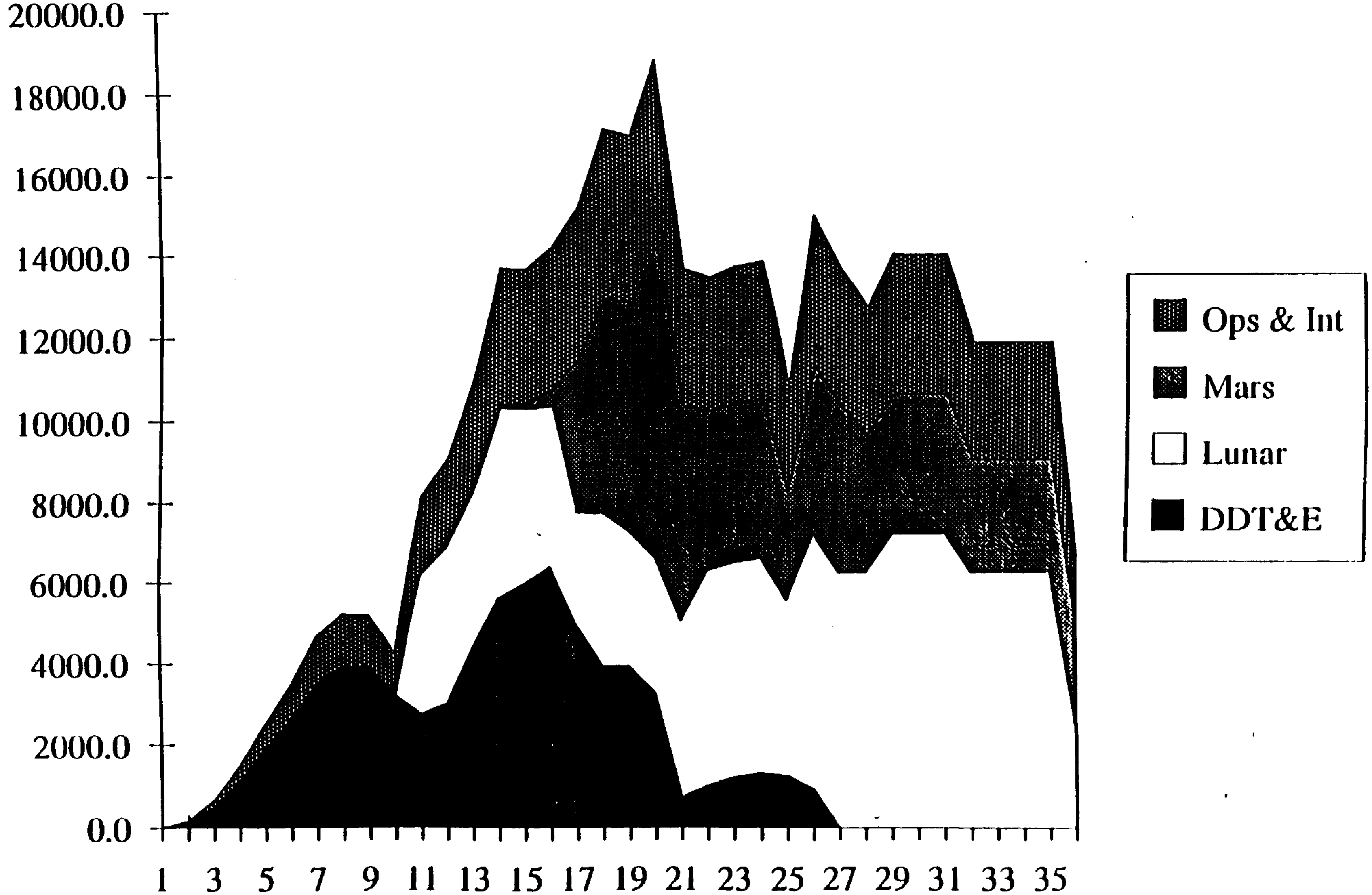
Industrialization and Settlement Cost Spreads

Our maximum scenario involved simultaneous industrialization of the Moon and progress towards settlement of Mars. As the cost spread shows, this is clearly beyond the funding levels recommended by the Augustine Commission. Both of the premises of this scenario, however, suggest significant private sector involvement.

What is significant in the result presented here is that investment on the order of $100 billions over about 20 years stretches from a plausible public-sector program of science and exploration to a program also involving the private sector for industrialization and settlement. This amount of funding is more than the private sector investment in the Alaska oil pipeline by a factor of a few, and probably less than the private investment in oil supertankers since the closure of the Suez Canal.

The economic potentials of lunar and/or Mars industrialization and settlement are presently not at all understood. We have made some stabs at estimating the costs. We have little or no idea as to the eventual payoffs.

Settlement/Industrialization Baseline w/Ops Int



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Results of Return on Investment Analyses

The facing page summarizes results of return on investment analyses. (The ROI methodology is explained in the technology and programmatics section of this briefing book.) Results designated "no ROI" had one case always more expensive than the other. An ROI can be calculated only when funding streams cross.

The storable case has very negative ROI because while less (i.e. no) technology money is spent, more vehicle stages must be developed so that the negative cost impact of not doing the essential cryo management and engine technology is large and early. The case for reusable lunar transportation is negative for a minimum lunar program and weak for a median program; it is strong for an industrialization-class program.

The other results were discussed earlier and are included here for completeness.

Return on Investment Analysis Summary

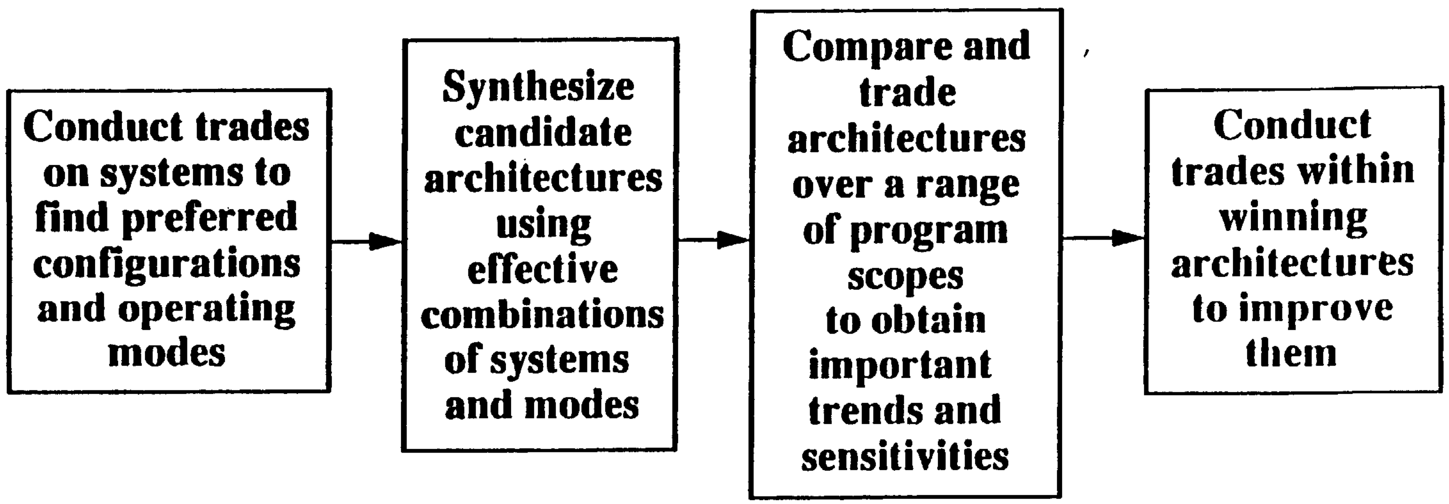
Case	Stor LOR vs cryo direct exp	Reus. LOR vs cryo direct exp	Reus. MEV vs exp MEV	SEP vs NTR $100/w	SEP vs NTR $500/w	SEP vs NEP	NTR vs cryo all-prop	NTR vs cryo all-prop	NTR vs cryo aero brake	NTR dash vs NEP	Lunar oxygen	Lunar oxygen
Program	Min	Full science	Ind/ settl	Full science		Any	Min	Full science		Ind/ settl	Full science	Ind/ settl
Result	-85	4.9	No ROI	9.6	-11	No ROI	1.7	15.9	13	44	4	10
Conclusion	Cryo	Reuse case weak	Reus MEV higher LCC	SEP	NTR	SEP better if less cost	CAP	NTR		NTR	No LLOX	LLOX

Strategy for Architecture Synthesis

The strategy we have adopted is illustrated on the facing page. First, we examined propulsion systems options through trade studies to understand how they work and to define preferred configuration operating modes. Secondly, based on the knowledge gained through these trade studies we chose a set of architectures using combinations of systems and modes, paying attention to integration compatibility, evolutions and commonality. Third, we will compare and trade architectures over a range of scopes and obtain important sensitivities and understand how architectures respond to program scope. We expect this analysis to lead to preferred architectures for various scopes. The final step is to conduct trades within the winning architectures to make further improvements.

All of this is guided by knowledge of the architecture cost drivers described earlier and by the knowledge gained on how systems work together, from the trades conducted within individual propulsion systems.

Strategy for Architecture Synthesis



Criteria:

Risk
Performance
Cost
Flexibility

Criteria:

Integration compatibility
Evolution
Commonality of hardware & technology

Criteria:

Ability to satisfy program goals & schedules
Cost

Criteria:

Risk
Performance
Cost
Flexibility

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Architectures Synthesis vs Mission/System Analysis

The facing page compares this approach to the traditional top-down systems engineering approach. The traditional approach shown on the right, starts with program goals, establishes mission requirements through trades, and continues to lower levels. As usually conducted, the traditional approach is faced with the great number of possible combinations noted earlier. The usual outcome is that requirement decisions are made and systems selected without trade studies.

The synthesis technique, on the left, attempts to avoid this problem by a combined top down/bottom up approach. It is similar to a classical optimization problem.

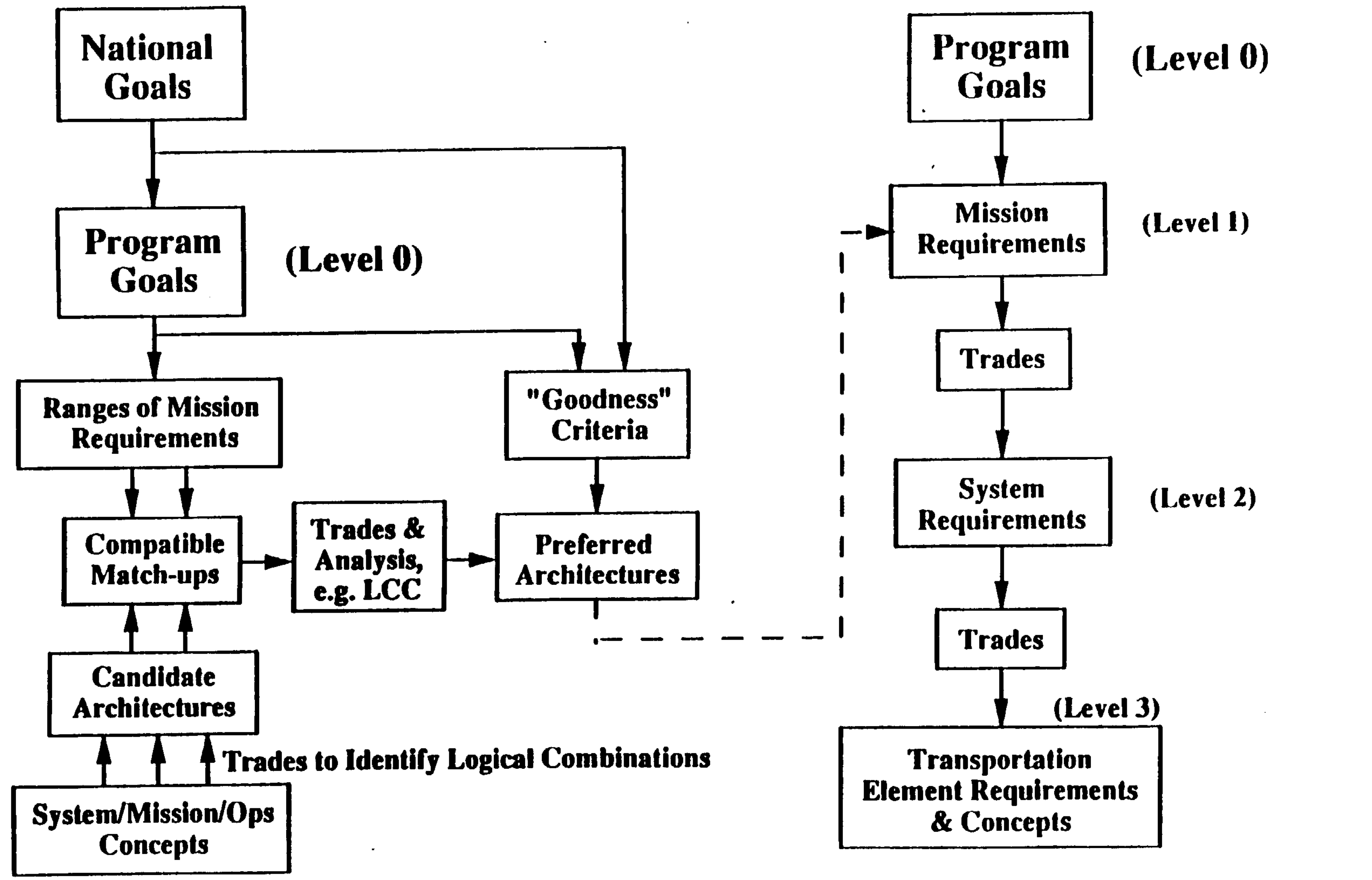
Optimization deals with infinite numbers of paths that satisfy boundary conditions. Optimization is a technique for generating only optimal paths. Any path that satisfies the boundary conditions is the sought optimal path.

Nothing quite as rigorous can be done in architecture synthesis. However, by bottom up trades, assembling systems into "good" candidate architectures, and matching with ranges of program scope, we may come close. The key is knowledge we obtain on what works well what things are compatible and combine well to satisfy mission requirements.

The last step is to conduct trades and analyses such as life cycle cost to identify preferred architectures, apply criteria derived from national goals program goals, to select among preferred architectures.

The dotted line indicates that one could then enter the traditional analysis flow with preferred architectures and their associated requirements and mission profiles, to further refine systems through systems engineering.

Architecture Synthesis vs. Mission/System Analysis



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Architecture Trade Flow

The facing page shows the low level system mission and operations trades that have been conducted or are being conducted for our seven architectures to represent the range of possible architectures for the SEI mission. Most of the trade areas have been presented in this briefing or have been presented in earlier briefings. The knowledge base in this area is fairly complete except that only very preliminary analyses have been done for the cryogenic direct mode and for cycler orbits. When these two options are completed we will be ready to finish up the architecture analysis.

Architecture Trade Flow

Cryo/Aero-braking	Nuclear Electric (NEP)	Solar Electric (SEP)	Nuclear Thermal Rocket (NTR)	L2/Lunar Oxygen	Cryo/Aero braking Direct	Cycler Orbits

• Mission design
• Reuse
• Aerobrake
  • shape
  • heating
  • GN&C
  • structures
  • assembly
• All-propulsive conj. option
• Modularity & commonality
	• Mission design
	  • trip time
	  • gravity assist
	  • node location
	• Power cycle
	• Power level
	• Specific power
	• Redundancy mgmt.
		• Mission design
		  • trip time
		  • gravity assist
		  • node location
		• Solar cell type
		• Power level
		• Specific power
		• Assembly/deployment of large space structure
			• Mission design
			• Isp and T/W sensitivity
			• Reuse
			  • tanks
			  • engines
			  • core stage
				• All-propulsive conj. option
				• Lunar oxygen benefits
				• Integration of lunar & Mars ops.
				• Advanced propulsion for LEO-L2 operations
					• Performance vs. separate MTV/MEV
					• Sensitivity to propellant choice
						• Mission design
						• Feasibility of high Mars encounter velocities
						• Design of "taxis"
						• Operational integration

For all: Overall configuration; key subsystems performance; integration compatibility; operations analyses

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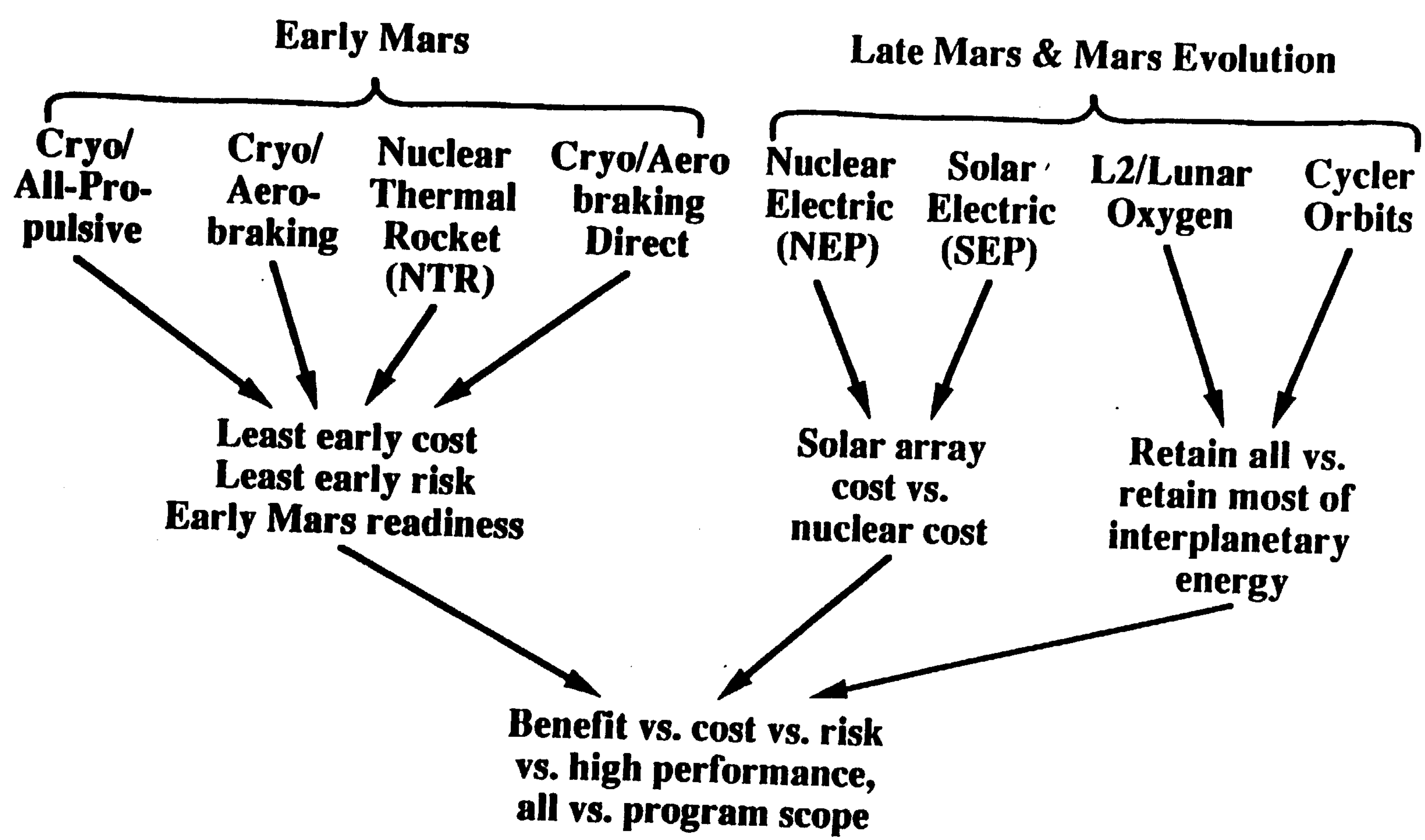
Architecture Evaluation Approach



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Mars Summary

- More than 20 beneficial modes identified.

- Early Mars: Cryo all-propulsive (CAP), ECCV*, conjunction;
  NTR all-propulsive, conjunction or opposition;
  Cryo aerobraking opposition, ECCV;
  (possibly) Direct with Mars oxygen.

- High performance, late Mars or evolution:
  SEP or NEP;
  ISRU, moon or Mars or both;
  Combintations.

- Efficiency range 10:1 measured as RMLEO (resupply mass LEO).

- Reusable MEV/Mars propellant has significant leverage for high-performance options.

* Earth Crew Capture Vehicle, an Apollo-like capsule used for Earth entry and landing or aerocapture to LEO. The rest of the vehicle is expended.

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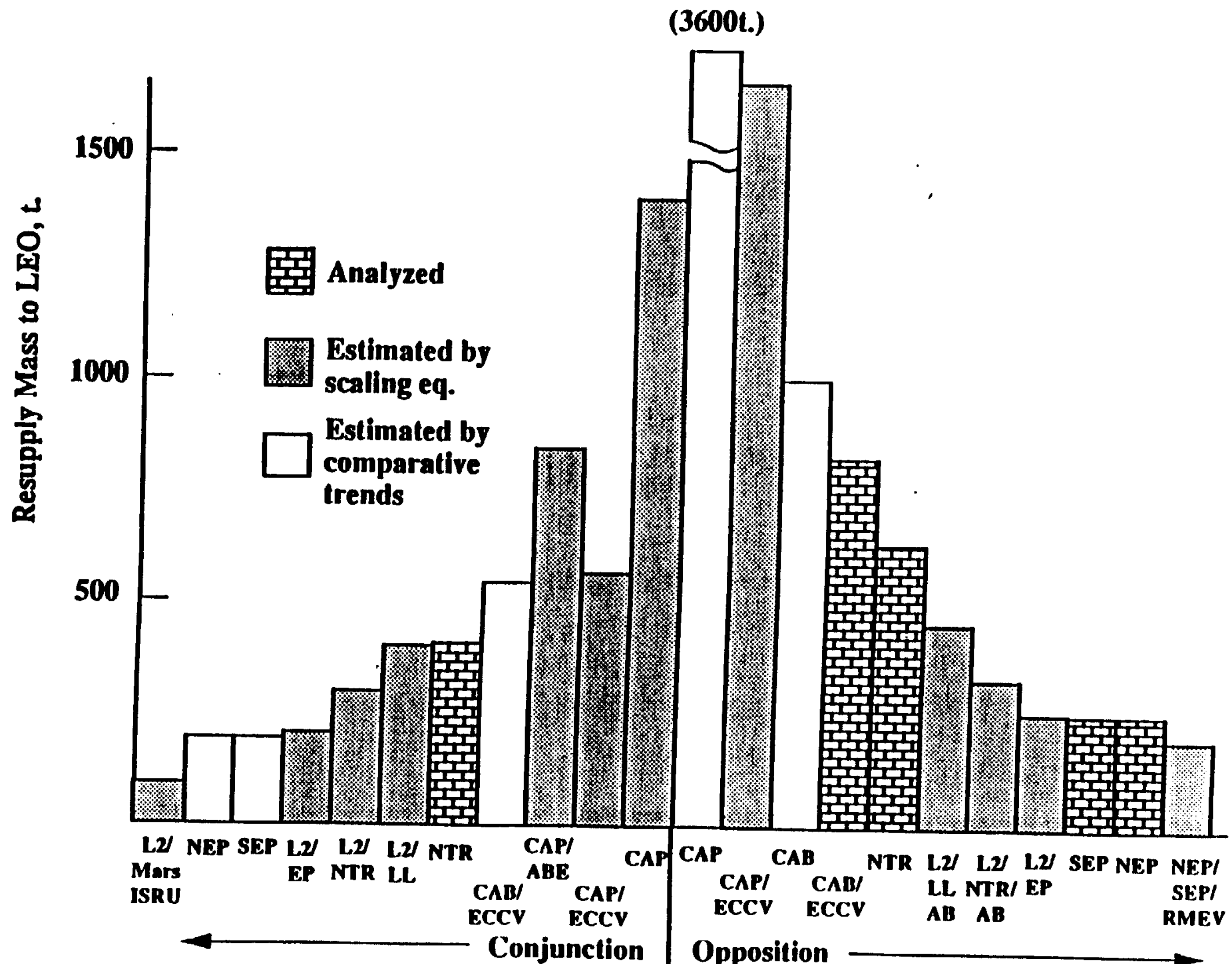
Comparison of Propulsion Options



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Conjunction vs. Opposition Mars Profiles

Opposition Advantages

- Shorter overall trip time, by at least a year.
- Transfer vehicle usually returns in time to be reused on next opportunity.
- Enables crew rotation/resupply mode with synodic period stay time.

Conjunction Advantages

- Lower energy; significantly less RMLEO unless very high Isp available.
- Venus swingby complexity not necessary.
- Long stay times at Mars.
- Shorter transfer times.
- Elliptic parking orbits can be optimized.

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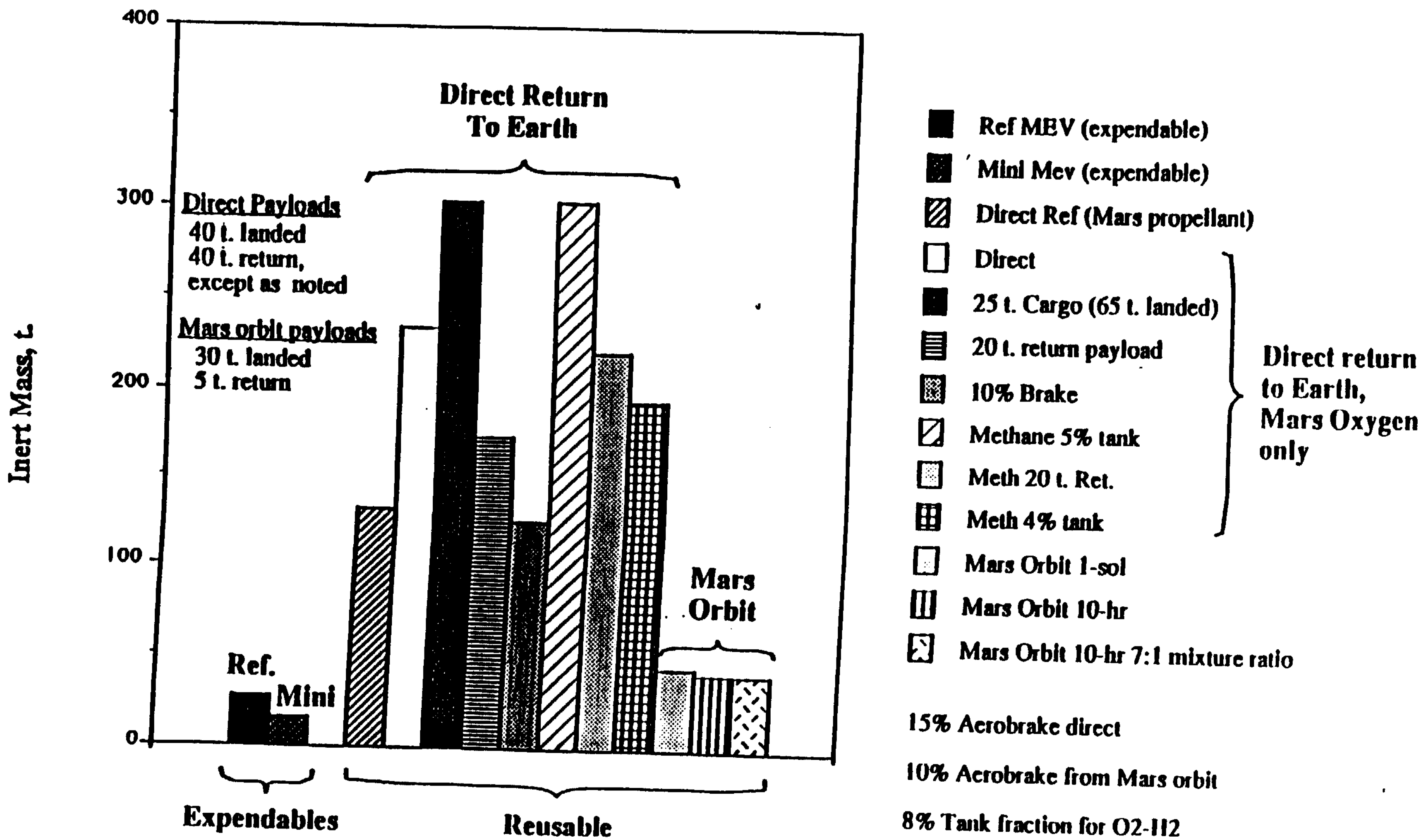
Reusable MEV Sensitivities



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II. Requirements, Guidelines and Assumptions

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Reference and Alternate Missions

Note: Contains material formerly in Mission Analysis

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Mission Analysis

A reference mission profile for Mars transfer was provided by MSFC, called the Level II reference case (year 2015 opposition opportunity). We investigated this profile for other opportunities in other years and did not limit ourselves to opposition mission only. An alternative mission profile is to use a direct transfer to Mars, refueling on the Martian surface, and direct return to Earth orbit (Mars Surface Rendezvous). A third alternative is to use the Earth-Moon Lagrange point two (L2) as a departure and return node.

The reference mission profile for the year 2015 depart on May 22nd of that year and has a 30 day stay time at Mars. The total mission duration is 565 days. A 2016 profile has a shorter overall time, 434 days, but adds about one kilometer/sec to the departure velocity change (ΔV) relative to the 2015 mission. The stay time at Mars is held to 30 days. Mission opportunities from 2010 to 2024 are tabulated. A plot of departure date versus outbound trip time for the 2013 opportunity with a 400 day stay is typical, showing a single optimum combination that minimizes mission ΔV. The other orbital characteristics than epoch of departure have some effect on mission velocity requirements such as capture and departure S-vector positions, GN&C maneuvers, etc. They have been investigated in as great a detail as the depth and length of the contract allows.

Optimum departure vectors indicate that the ability of the engines to be capable of multiple burns and therefore do broken plane trajectories to the declination launch asymptote will be a requirement. Arrival conditions at Mars for capture orbit parameters such as periapses location and lighting (capture and land in light), impacts of true anomaly and parking orbit period on the relative position of the S- vector (departure vector) for abort and departure capability and into the characteristics of the aerobrake itself for GN&C, landing and crossrange capability.

In the reference mission, the excursion vehicle and transfer vehicle separately capture into Mars orbit. To allow a one day spacing between the captures, a velocity difference must be generated between them. To keep this ΔV low (under 100 m/s), the separation should occur about 50 days before Mars arrival.

Guidance, navigation, and control analyses can be done at different levels of detail from closed form approximations to full 6-degree of freedom simulations. To date 3 degree of freedom analyses with variable atmosphere has allowed assessment of the errors induced by a variable atmosphere. Guidance laws are being investigated.

Aeroheating analyses were performed on the medium (L/D=0.5) and higher (L/D=1.1) lift brake concepts. With a fixed exit velocity target, flying inverted (negative L/D) extends the time for the maneuver. This is because the maneuver is from a hyperbolic velocity. A zero lift trajectory would rise quickly back out of the atmosphere. Negative lift 'holds down' the vehicle, extending the time for the aeromaneuver, thus lowering the heating rate. The heat rate itself varies widely depending on the analysis method. This is an area that requires more detailed investigation in the future. Either method leads to peak temperatures at the stagnation point in excess of 2000K for hyperbolic excess energies (C3) of over 30 km^2/sec^2. Aeodynamic loads were estimated over the brake surface, and two structural concepts were examined. The first was a spar framework, the second was a truss framework.

A summary of this work is given below:

This summary addresses the aerobrake analyses categorized as geometric configuration for capture and landing, Mars atmosphere knowledge uncertainty impacts on GN&C, design configurations for reducing heating rates and loads, landing flight mechanics for range and crossrange requirements, structural techniques for reducing weight, and integration of technology to meet overall mission goals. The aforementioned categories will be covered in four sections: Aerocapture, Heating, Structure, and Ascent/Descent.

Aerocapture - Critical GN&C related aerocapture issues are line-of-apside control and apoapsis altitude control. Aerocapture analyses results included in this summary show the following:

* Asymmetric roll with a finite rate provides improved line of apsides control.

* A guidance system designed for a low density atmosphere needs to be optimized for other atmospheric conditions.

* Using MarsGram, a one sigma density change results in a large difference in density variation between day and night.

* The guidance system (as related to aerocapture exit conditions) is more effected by large (wavelength > 1000 km) horizontal sine wave density variations.

* A larger vertical wavelength (on the order of 20 km) sine wave density induces a lesser error than a smaller vertical wavelength (on the order of 5 km) sine wave density.

Heating - Mars aerocapture heating analyses results are given for stagnation point heating and for some choices of surface heating. Heating analyses results included in this summary indicate the following:

* For the Mars aerocapture MTV, the stagnation point heating rate resulting from averaged lift-down L/D is lower than the heating rate for average lift-up L/D.

* Under similar conditions, the heating loads follow the same trend as the stagnation point heating rate.

* Along the center streamline of the hyperboloid aerobrake the predicted radiative heat transfer rate at Mars using the Park method is approximately two times that using the Tauber-Sutton method.

* The total heating rates at the stagnation point with Park(146 w/sq cm) and Tauber-Sutton (80 w/sq cm) are higher than the near term (1993) radiative material capabilities of approximately 70 w/sq cm.

* For an averaged L/D = 0.5 the stagnation point heating rate for Mars aerocapture is 146 w/sq cm; Earth aerocapture heating rate is 172 w/sq cm.

* The local Reynolds number along the aft streamline of the 30m body does not exceed 10E6.

Structures - Structural analyses results demonstrate weight savings and strength improvements through advanced composites application and through spar design advantages. Included structural analyses results depict the following:

*Spar and truss configurations were developed for the 30 meter aerobrake concept.

* For the spar configuration and with current technology, the (81 mt payload) weight estimate is 41.5 klb and the MTV (153 mt payload) estimate is 66.3 klb.

* Improved material characteristics (200 ksi vs 105 ksi span strength) reduces configuration weight by greater than 15%.

* Mass savings of 30% may be achieved by improved spar design and advanced materials characteristics.

* The truss configuration provides a 15% weight savings compared to the spar configuration.

Ascent/Descent - No ascent related information is discussed in this version of the IP&ED; a forthcoming update will contain a discussion of ascent related data.

Descent trajectory analyses results point to L/D requirements related to landing site accessibility issues. Included descent trajectory results include the following:

* For MEV with L/D = 1 and descent inclination of 45 degrees, a displacement in latitude of 30 degrees may be achieved.

* An increase in L/D from 0.5 to 1.0+ extends the range by approximately 50%.

* An aeroflare reduces the ideal delta velocity required for landing by 200 to 300 m/sec (L/D = 1).

* Cross range is a function of L/D with atmospheric density and dust concentrations affecting the results

Issues with large aerobrakes such as these center around on-orbit assembly and inspection, functions which consume many man-hours for the Space Shuttle on the ground. The Shuttle has the only reusable aerobrake with repetitive use and accessible data. Another issue is selection of the landing site. If the landing site requires an extensive plane change, the L/D is higher, which ripples through the packaging of the lander and the weight of the lander back to Earth launch requirements. On the other hand, using an arrival orbit tailored to the landing site also has an impact on propulsion requirements, thence to Earth launch mass. Thus selection of a landing site is required early since it affects the whole design in a complex way. Some candidate sites are listed. Either a reference site or a requirement to meet a range of sites up to some level of difficulty (for example any site less than +5km altitude and <70 degrees latitude) needs to be given as an input requirement for further analyses.

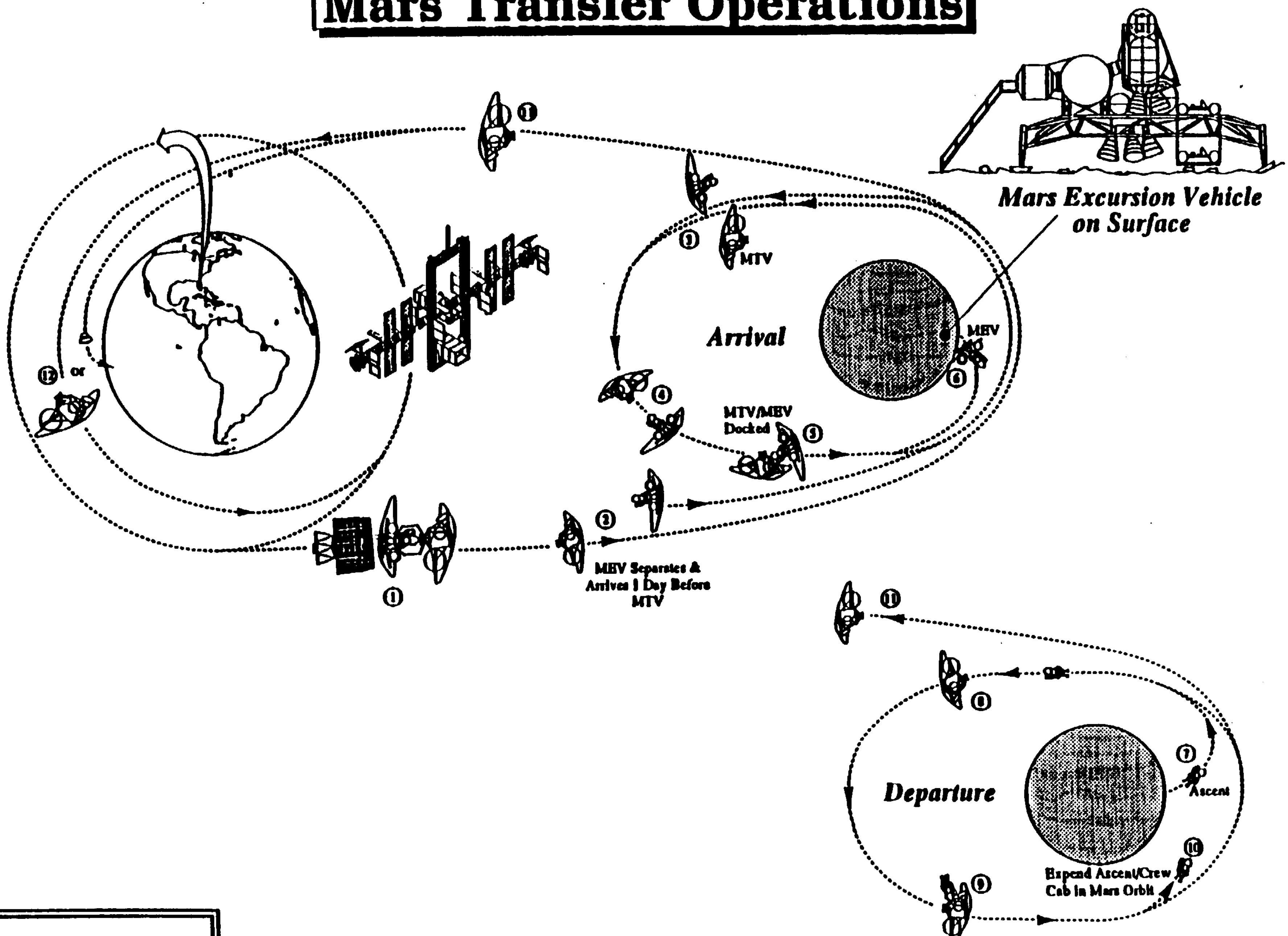
Mars Transfer Operations

Mars transfer operations for the reference system are illustrated here. The opposition profiles normally include a Venus swingby either going to Mars or returning to Earth. Occasionally, a Venus swingby may be used each way. (The reference 2015 mission uses an outbound swingby; there is also an alternate inbound swingby profile for this opportunity.) Venus swingbys are normally unpowered and there are no operational events at the swingby. The nominal mission sequence is as follows:

1. Reference Cryo/AB Mars vehicle leaving Earth orbit.
2. MEV/MTV separate 50 days from Mars.
3. Unmanned MEV captures into Mars orbit 1 day prior to MTV.
4. MTV/MEV rendezvous and berth in Mars orbit.
5. Crew transfers from MTV to MEV.
6. MEV descends to the surface of Mars.
7. MAV ascends from surface, leaving descent stage.
8. MAV/MTV berth in Mars orbit.
9. Crew transfers from MAV to MTV.
10. MAV left in Mars orbit.
11. MTV departs from Mars toward Earth.
12. MTV captures in LEO, or crew returns to Earth's surface in ECCV.

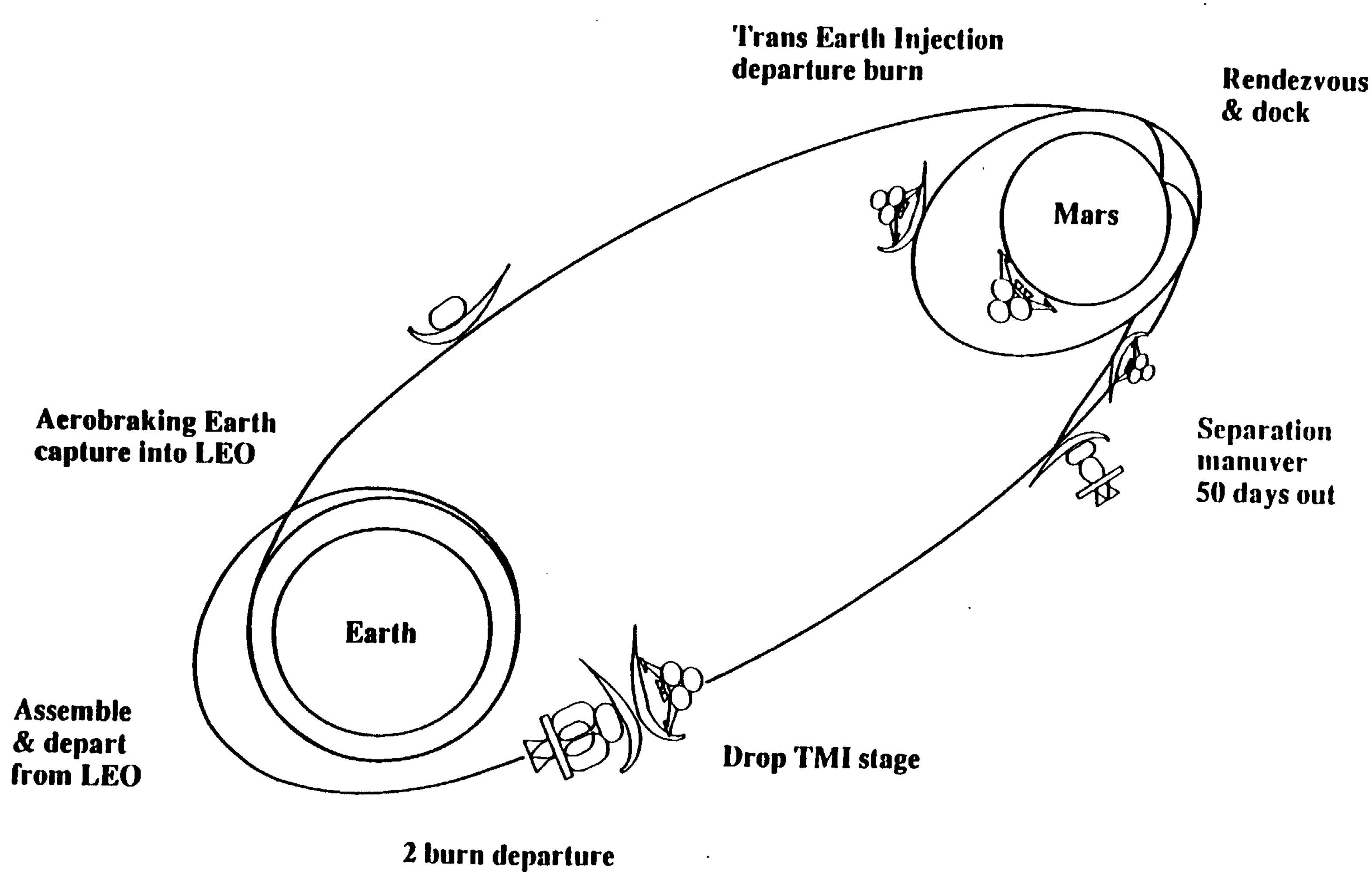
Acronymns: AB - aerobraking; MEV - Mars Excursion Vehicle; MTV - Mars transfer vehicle, includes trans-Mars injection stage as well as transfer propulsion and hab; MAV - Mars ascent vehicle; LEO - low Earth orbit; ECCV - Earth crew capture vehicle (like an Apollo command module).

Mars Transfer Operations



MSFC

Reference Chemical/Aerobrake Mission Profile Schematic



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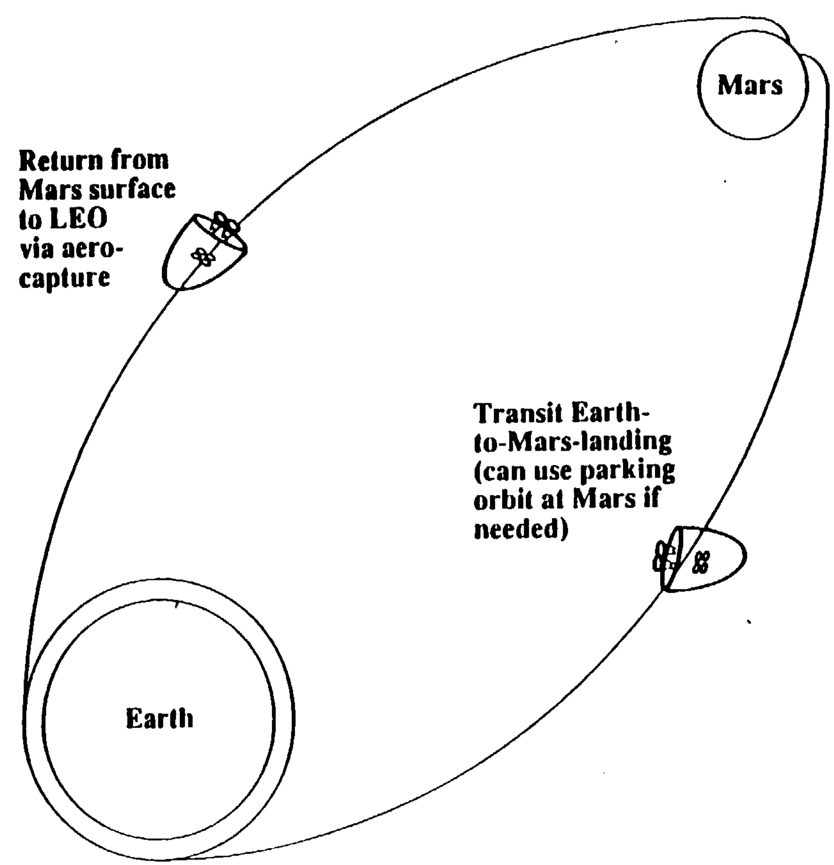
Mission Schematic for Mars Direct System (Conjunction Profiles, Refueled at Mars)

Delta V Mars to Earth (m/sec):

Ascent to circular	4100
Injection C3 = 10	2373
Earth aero-capture	200
Reserve	120
Total	6793



Delta V LEO to Mars landing (m/sec):

TMI (C3 = 20)	4100
Midcourse & aerocapture	200
Landing	1500
Reserve	115
Total	5915

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Earth-L2-Mars Mission Profile Schematic

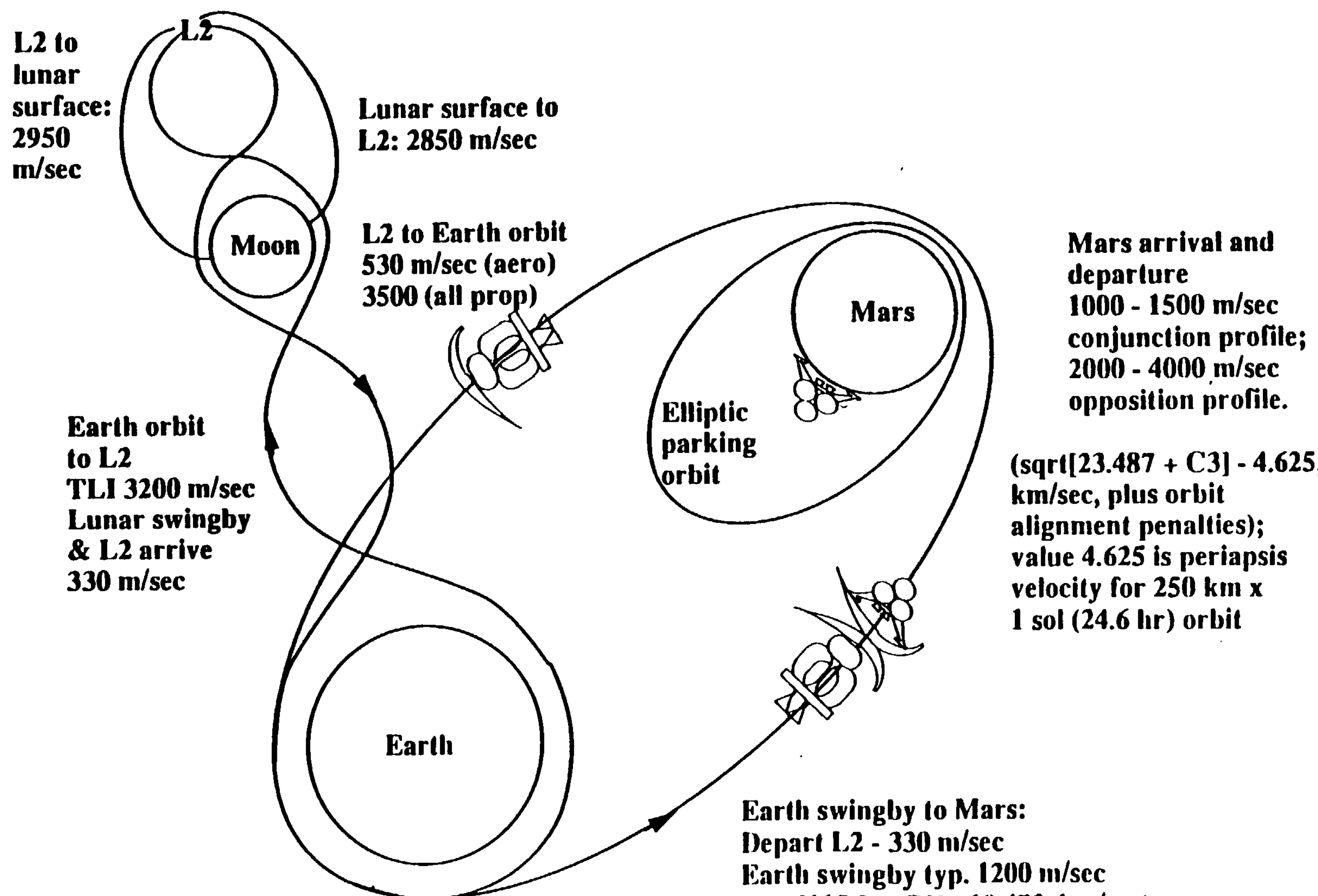
The use of the L2 libration point as a transportation node was originally suggested by the Farquhar and later by Keaton. The L2 node has three advantages. (1) the delta V from L2 to Mars transfer is almost 3000m/sec less than from low Earth orbit. As a result the vehicle is much smaller; (2) the transportation node is not in the low Earth orbit debris environment; (3) the L2 node is a suitable location for using lunar oxygen in Mars mission systems.

The L2 node scheme involves a network of transportation from the Earth to L2 and back, from L2 to the lunar surface and back, and L2 to Mars and back. The facing page indicates the delta V's for each leg of the transportation network.

Transportation to the surface of the moon via L2 is efficient when lunar oxygen is used for the LEV, as efficient as the low lunar orbit node with lunar oxygen. (The LEV becomes like the LTV in size). The disadvantage of the L2 node is that substantially greater lunar oxygen production is required compared to the low lunar orbit node. The advantage is that any site on the lunar surface is accessible any time and return to Earth is available at any time. Delivery of lunar oxygen to the L2 node by rocket is surprisingly efficient.

Transfer from L2 to Mars uses dual powered gravity assist, at the Moon and the Earth. This means that launches to Mars are limited to times the Moon is at proper location. The launch window problem is very similar to launching from a Earth orbit. In either case the launch window problem is difficult for opposition missions but for the much longer launch windows of conjunction missions, multiple chances are available. The return from Mars uses aero-assist to Earth and powered gravity assist from the Moon to L2.

Earth-L2-Mars Mission Profile Schematic



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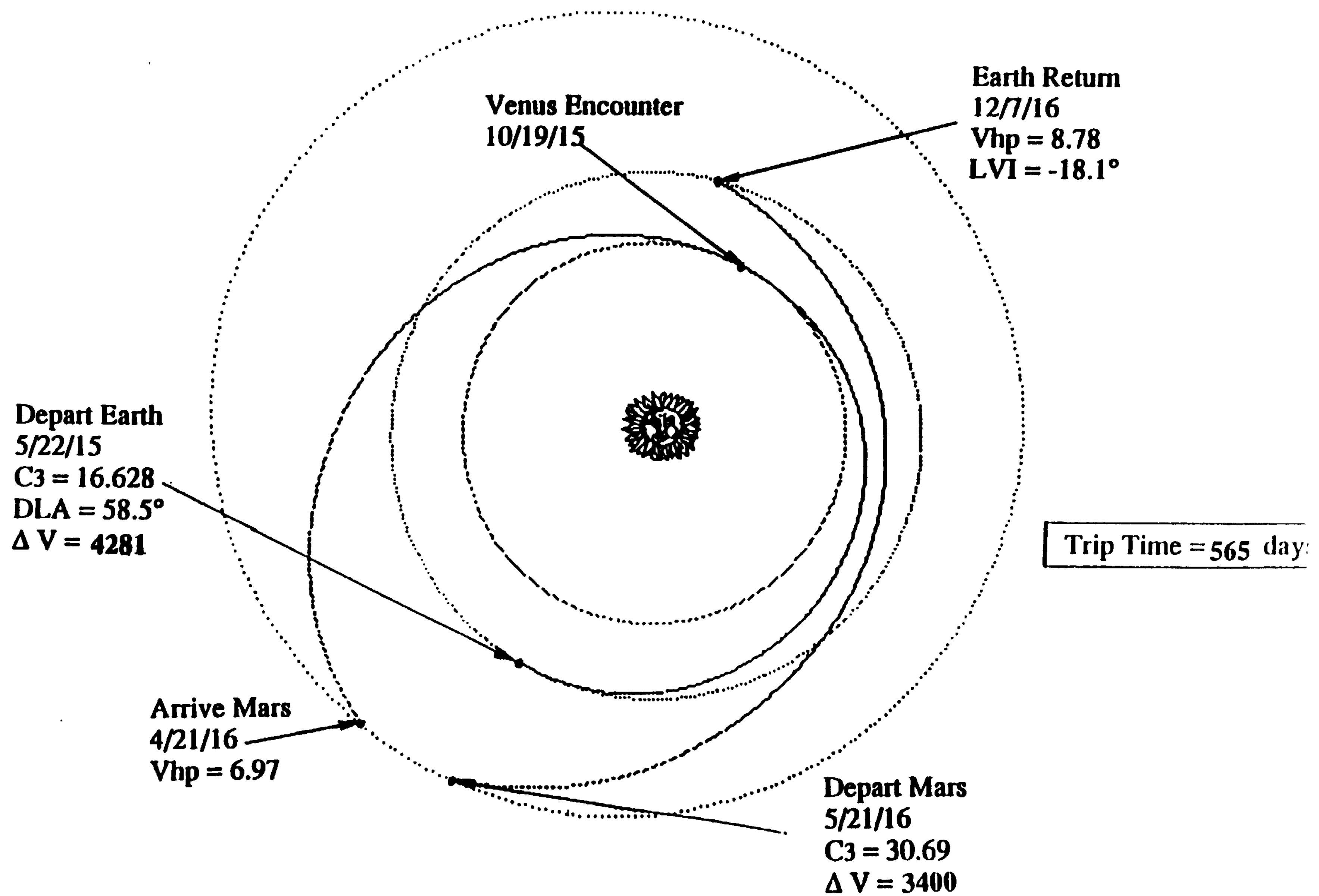
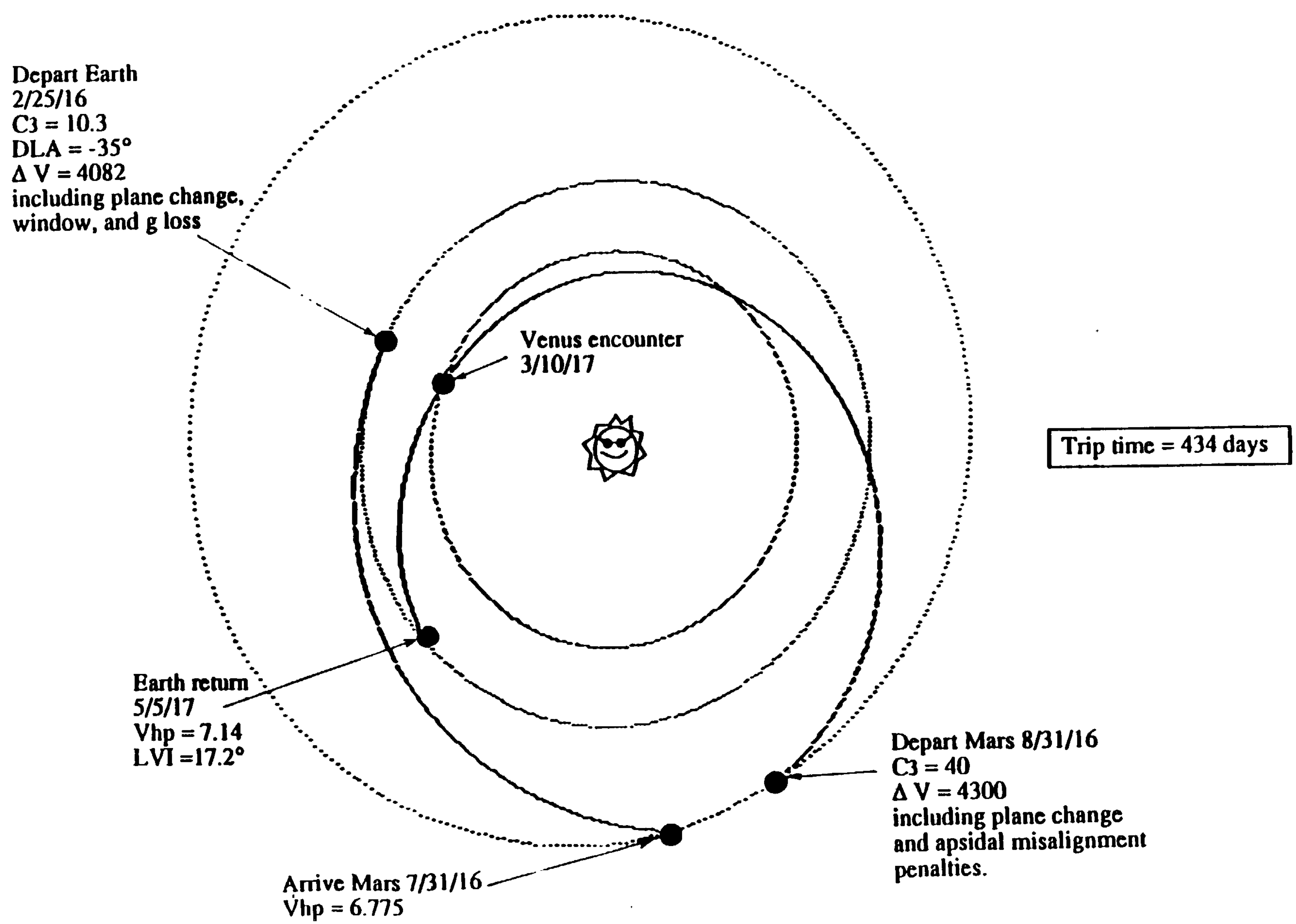
Reference 2015 Mission Profile



Boeing Case #2



Mars Trajectory Data

STCAEM/pab/10/19/90

Opportunity (optimized runs)		Earth Dep C3	Mars Arr. Vhp	Mars Departure C3	Mars Departure Δ V	Earth Arrival Vhp	Earth Arrival C3	Mars Orbit Inc. (Deg)	Earth launch DLA	Mars Arr. LVI
2010 Opposition	12/01/10 -10/26/11, 11/25/11- 8/31/12	28.69	4.93	16.1	2.32	6.99	48.9	30	4.19	-28.63
2010 Conjunction	10/26/9 -10/31/10, 8/27/11 - 7/15/12	11.14	3.26	7.0	2.73	3.80	14.4	30	32.82	9.48
2013 Opposition	11/22/13 - 8/6/14, 10/5/14 - 8/14/15	13.08	4.10	37.7	3.54	4.53	20.5	30	20.42	25.32
2013 Conjunction	12/3/13 - 9/23/14, 9/28/15 - 9/6/16	9.58	3.15	6.9	2.61	4.47	20.0	45	23.73	31.13
2015 Level II Reference	5/23/15 - 4/22/16, 5/22/16 - 12/8/16	20.19	7.02	30.0	4.70	8.77	77.91	30	55.92	14.51
2015 Level II Alternate	10/15/15 - 7/16/16, 8/15/16 - 5/17/17	48.38	4.79	33.3	3.58	3.94	15.5	30	0.18	15.3
2015 Conjunction	12/24/13 - 11/17/14, 12/14/15 -10/8/16	8.89	4.22	5.4	2.37	5.52	30.5	35	18.45	32.93
2015 L II Ref. + 50 day	5/23/15 - 4/22/16, 5/22/16 - 1/27/17	14.19	6.93	18.0	2.37	9.47	89.7	30	55.92	15.22
2016 Boeing Nominal	2/25/16 - 7/31/16, 8/31/16 - 5/5/17	10.34	6.82	39.7	4.37	7.14	51.0	30	-35.94	-1.69
2018 Opposition	3/27/17 - 3/10/18, 4/24/18 - 12/18/18	19.71	5.96	10.9	2.58	5.04	25.4	30	17.97	22.3
2018 Conjunction	5/12/18 -11/28/18, 5/31/20 - 11/27/20	7.86	2.97	7.9	3.41	4.02	16.2	45	-37.94	-7.81
2020 Opposition	6/4/20 - 12/11/20, 1/10/21 - 1/28/22	24.40	3.89	27.8	3.77	4.28	18.3	20	15.1	-9.69
2020 Conjunction	7/20/20 - 1/16/21, 8/09/22 - 1/16/23	13. 40	3.13	18.8	3.92	6.67	44.5	30	18.65	-3.40
2022 Opposition	11/11/21 - 9/17/22, 10/17/22 - 6/5/23	16.31	5.31	43.2	4.46	6.00	36.0	20.	-60.02	-4.04
2023 Conjunction	9/8/22 - 4/16/23, 7/9/24 - 5/5/25	19.03	3.18	12.3	2.66	2.86	8.81	30	50.73	23.71
2024 Opposition	9/20/23 - 7/4/24, 8/4/24 - 5/30/25	27.91	6.46	9.0	1.61	3.08	9.49	30	-19.58	-6.53
2025 Conjunction	10/17/24 - 6/24/25, 8/11/26 - 5/5/27	19.68	3.00	8.3	2.60	2.60	6.76	35	55.88	34.75

Level II Reference : ΔV Earth Departure = 4281 m/sec ΔV Earth Arrival = 6278 m/sec (at LEO)
Data From MASE ΔV Mars Arrival = 3949 m/sec ΔV Mars Departure = 3400 m/sec

ADVANCED CIVIL SPACE SYSTEMS

Trajectory Information for Mission Opportunities

BOEING

STCAEM/pth/19Mar90

Opportunity	C3 Earth Departure	Vhp Mars Arrival	C3 Mars Departure	Vhp Earth Arrival	Periapsis Altitude (km)	Apoapsis Radius (km)	Periapsis Radius (km)	Eccentricity	Semi-major Axis (km)
2010 Opposition	28.68	4.93	16.07	6.99	250	37188.13	3647	0.82	20415.57
2010 Conjunction	11.14	3.26	7.04	3.72					
2013 Opposition	13.07	4.10	37.68	4.53					
2013 Conjunction	9.58	3.15	6.88	4.47					
2015 Level II Reference	20.19	7.01	29.97	8.76					
2015 Level II Alternate	48.36	4.79	33.28	3.94					
2015 Conjunction	8.89	4.22	5.42	5.52	500	36932.86	3897	0.81	
2015 L II Ref. + 50 day	14.21	6.93	17.97	9.47	250	37182.86	3647	0.82	
2016 Boeing Nominal	10.34	6.82	39.72	7.14				0.82	
2018 Opposition	19.71	5.96	10.93	5.04		17243.37		0.65	10443.19
2018 Conjunction	7.86	2.97	12.28	4.02		37188.13		0.82	20415.57
2020 Opposition	24.39	3.89	27.83	4.28		36521.48		0.81	
2020 Conjunction	13.40	3.89	20.03	6.67		37188.13		0.82	
2022 Opposition	16.31	5.31	43.16	6.00		36521.48		0.81	
2023 Conjunction	19.03	3.18	9.31	2.86		37188.13		0.82	
2024 Opposition	27.91	6.46	9.02	3.08		37188.13		0.82	
2025 Conjunction	19.68	3.00	7.81	4.79	250	37188.13	3647	0.82	20415.57

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Orbit Insertion Lighting Angle, Latitude, and Approach Turning Angle for 2010 to 2025 Mars Mission Opportunities

Opportunity	Periapsis Lighting Angle (°)	Periapsis Latitude (°)	Approach Turning Angle (°)
2010 Opposition	54.19	1.21 N	70.96
2010 Conjunction	42.20	42.51S	58.30
2013 Opposition	21.94	24.33 S	65.70
2013 Conjunction	55.01	16.40 S	57.22
2015 Level II Reference	22.57	28.22 S	78.88
2015 Level II Alternate	35.45	29.97 S	70.19
2015 Conjunction	67.75	22.12 S	67.58
2015 L II Ref. + 50 day	23.27	27.96 S	78.67
2016 Boeing Nominal	11.15	28.88 S	78.37
2018 Opposition	36.53	24.21 S	75.60
2018 Conjunction	50.52	47.65 S	55.16
2020 Opposition	13.50	15.97 S	63.94
2020 Conjunction	10.67	22.59 S	57.01
2022 Opposition	66.41	26.90S	72.90
2023 Conjunction	10.61	1.99 N	57.50
2024 Opposition	68.29	26.75 S	77.31
2025 Conjunction	15.32	22.06 S	55.55

Data generated by the PLANET program, property of the Boeing Company.

2013 - 400 Day Stay -- 900 Days Round Trip

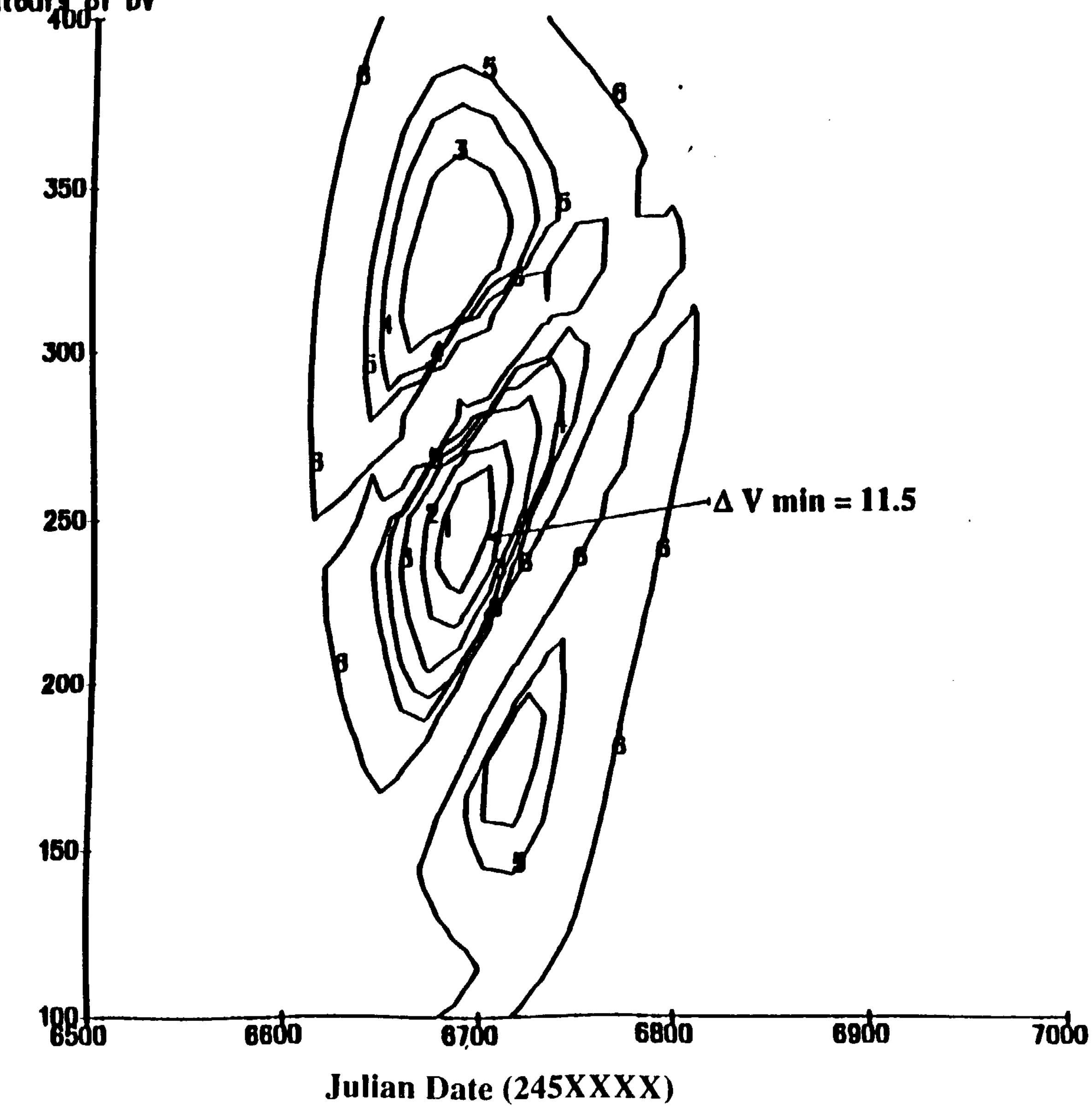
2013 Conjunction minimum round trip ΔV is approximately 11.5 km/sec with an outbound/inbound trip time of approximately 240 days.

/STCAEM/mk/12June90

900 Days Round Trip

2013 CONJUNCTION DELTA VEL SCAN
EARTH Departure, MARS Arrival
Contours of DV

Contour	ΔV
1	11.5
2	12
3	13
4	14
5	15
6	20



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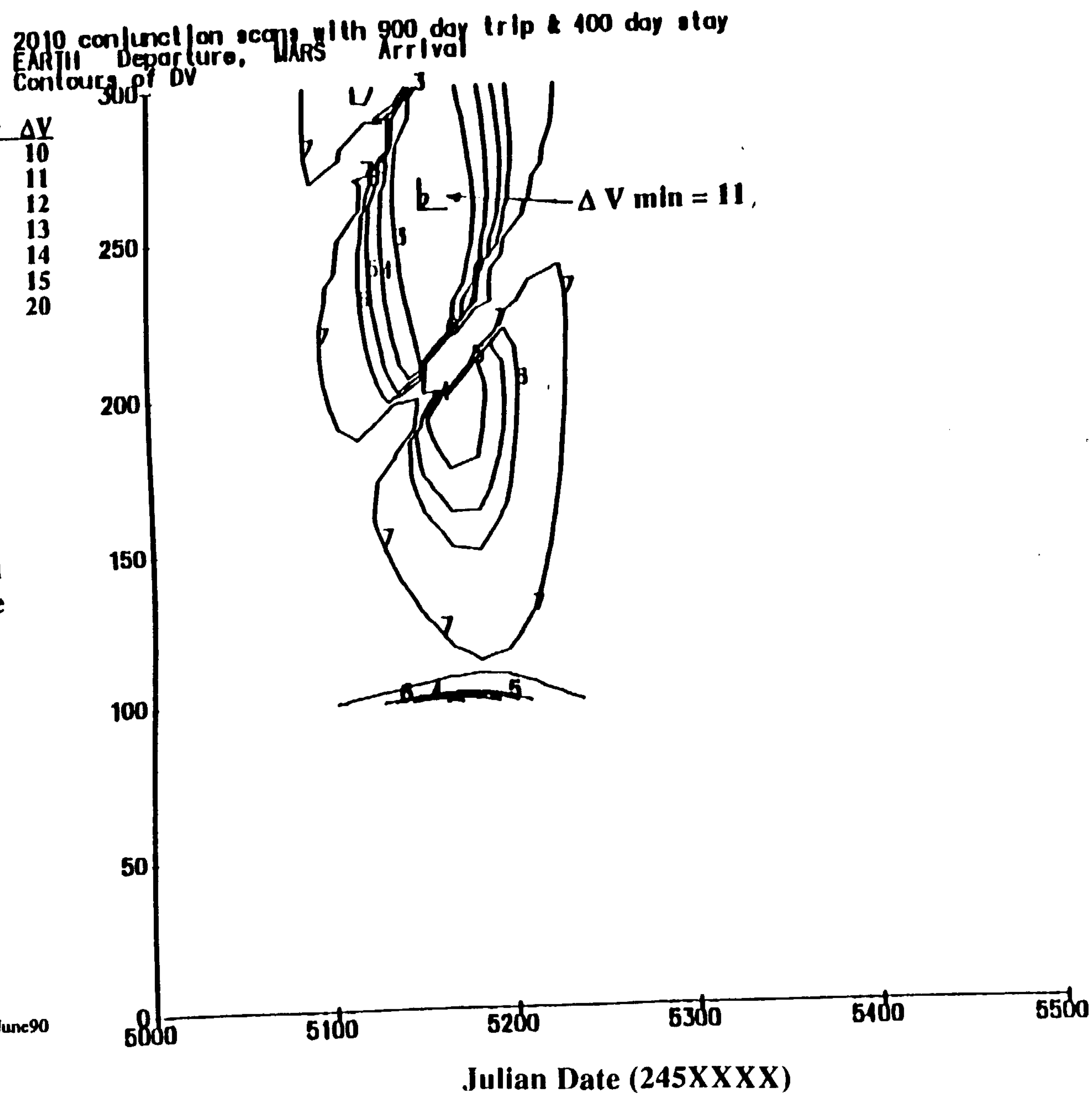
900 Days Round Trip

Contour	ΔV
1	10
2	11
3	12
4	13
5	14
6	15
7	20



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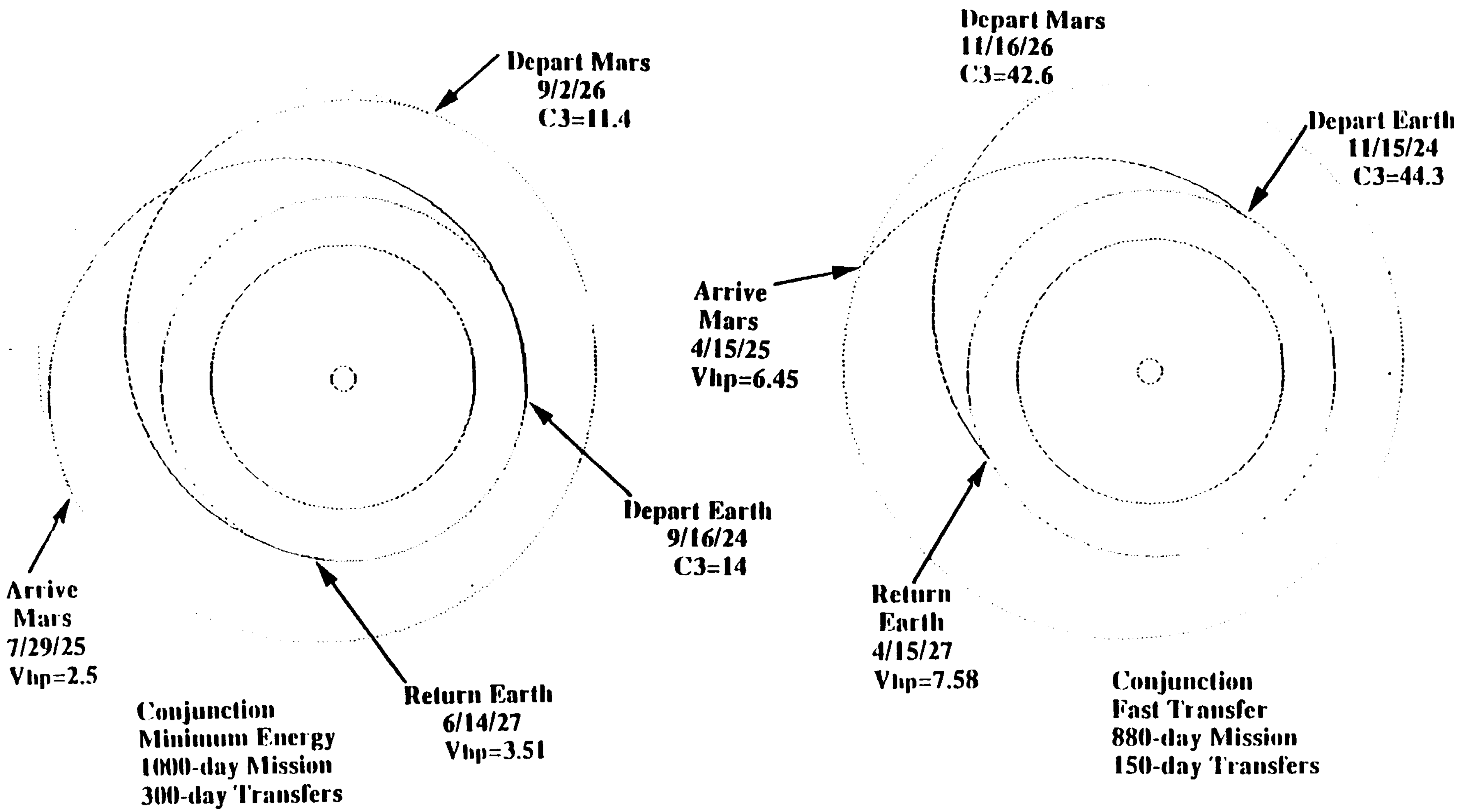
Conjunction Class Minimum Energy and Fast Transfer Missions

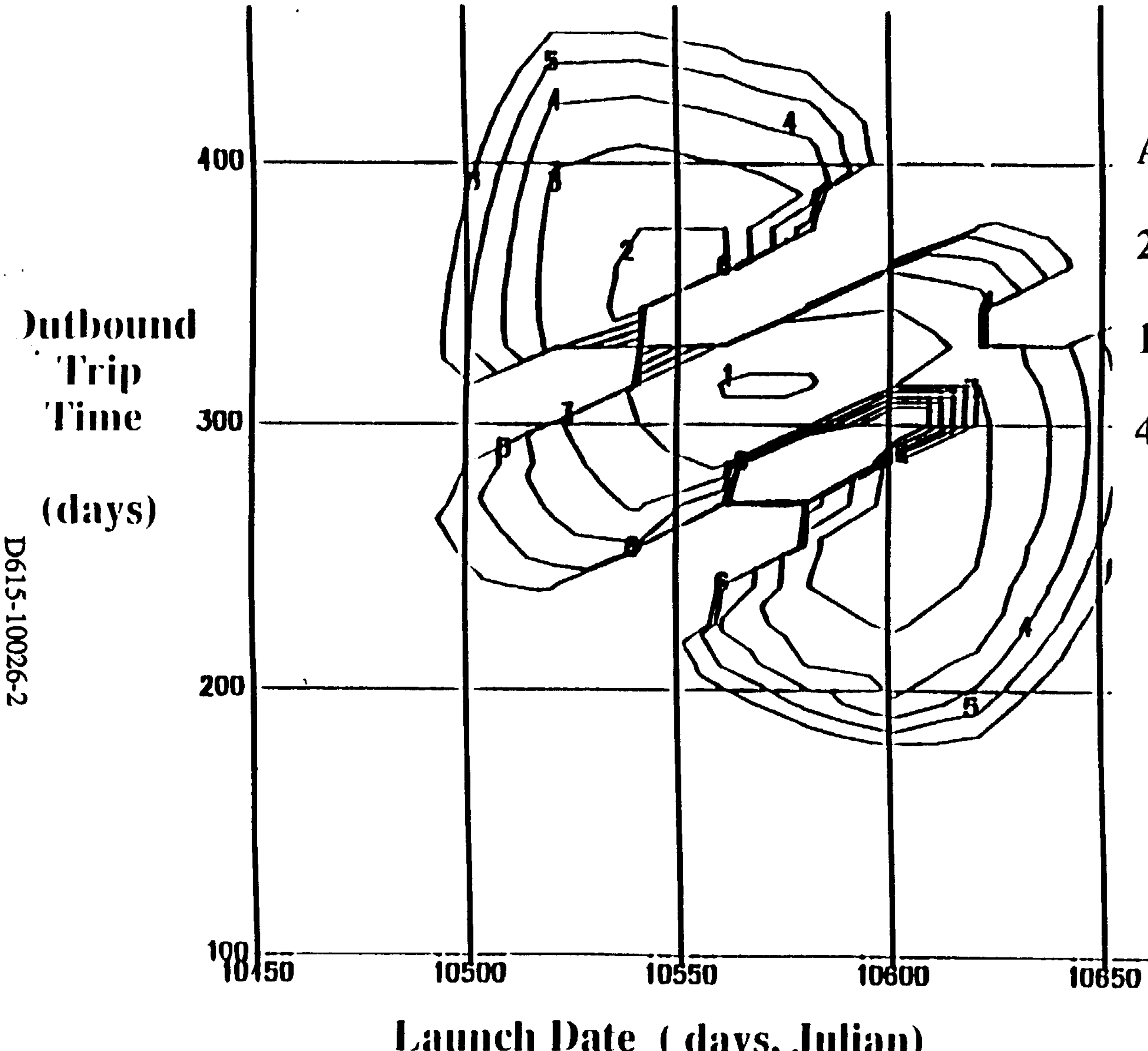
Shown in the next several charts are the data comparing the results of using a "fast transfer" versus a minimum energy (minimum ΔV) mission for the 2025 time frame. As shown a price in ΔV costs at all stages of the trip must be paid for the reduction of time spent in transit and the reduced risk of crew exposure to Galactic Cosmic Rays. This price will be reflected in the IMLEO of the vehicle.

For a range of trip times and launch dates, contour plots of total ΔV for both cases were generated. It is apparent from the data that the fast transfer missions have narrower lunch opportunities at a higher total ΔV price but with a transit time of half or less than a normal mission.

Minimum Energy and Fast Transfer



Delta-Velocity Contour
Typical Conjunction Mission



Analysis Tool-Boeing's PLANET

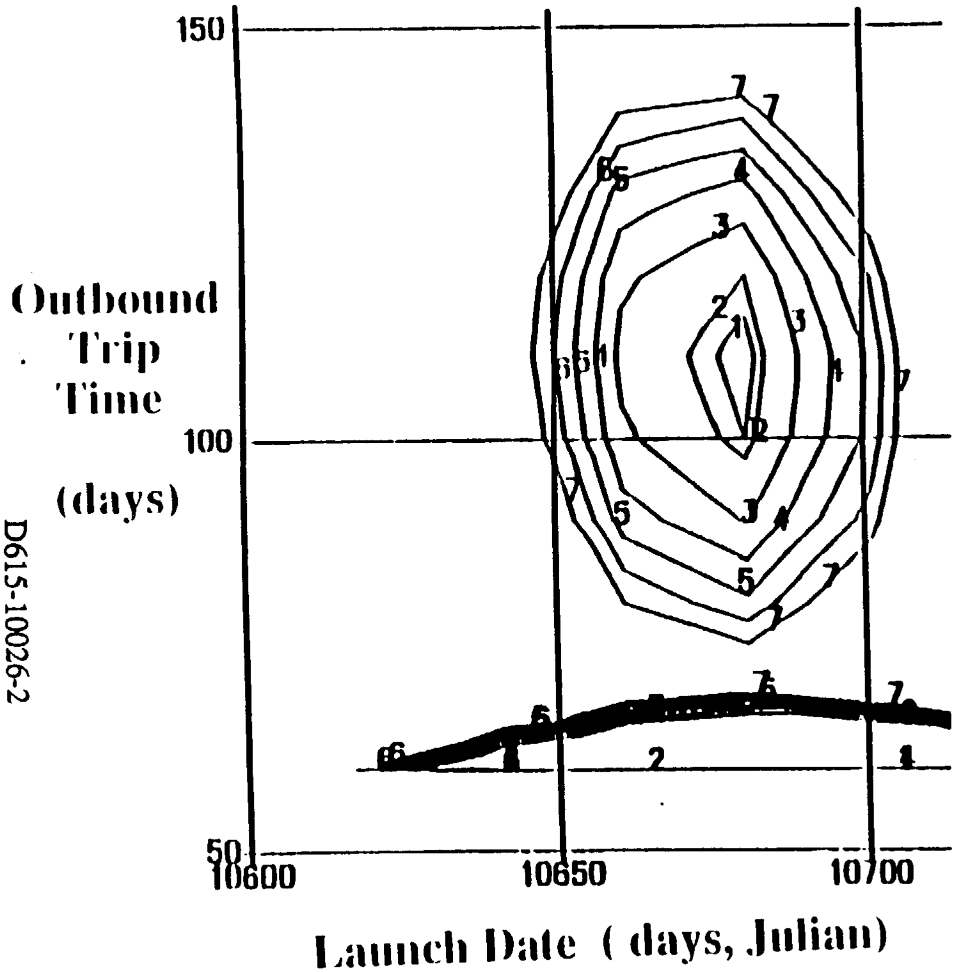
2025 Typical Conjunction Mission

1000 Day Total Trip Time

400 Day Stay at Mars

	km/s
1	5.3
2	6.0
3	7.0
4	8.0
5	9.0
6	10.0

Delta-Velocity Contour
Fast-Trip Conjunction Mission



Analysis Tool -Boeing Developed PLANET

2025 Fast-Trip Conjunction Mission

800 Day Total Trip Time

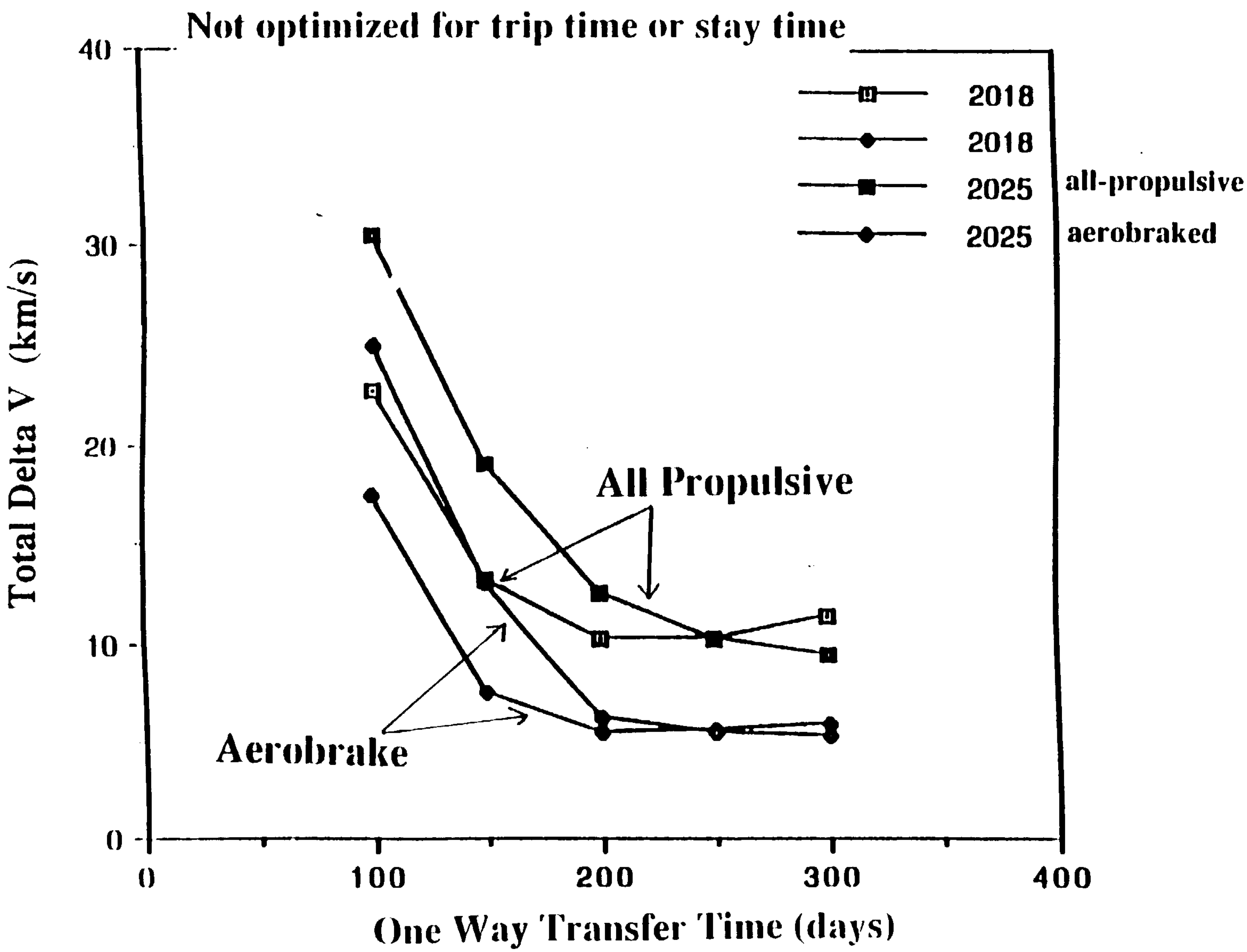
600 Day Stay at Mars

	km/s
1	30.7
2	31.0
3	32.0
4	33.0
5	34.0
6	35.0
7	36.0

Conjunction Fast Transfers Preliminary Delta Velocity Trends

This is a comparison of the total required ΔV versus transfer time one way for aerobraked and all-propulsive vehicles . This was done for two years; the fairly "easy year" of 2018 and the "hard year" of 2025. This shows both the effect of the range of years and the effect of using an areobraked vehicle. The aerobrake shows its advantage in the reduction in ΔV that must be provided. The advantage in going in 2018 is evident as long as the total transfer time is less than 250 days, beyond that the advantage is lost. However this data has not been optimized for the transfer time, and only shows the relative advantage of using an aerobrake.

Preliminary Delta Velocity Trends



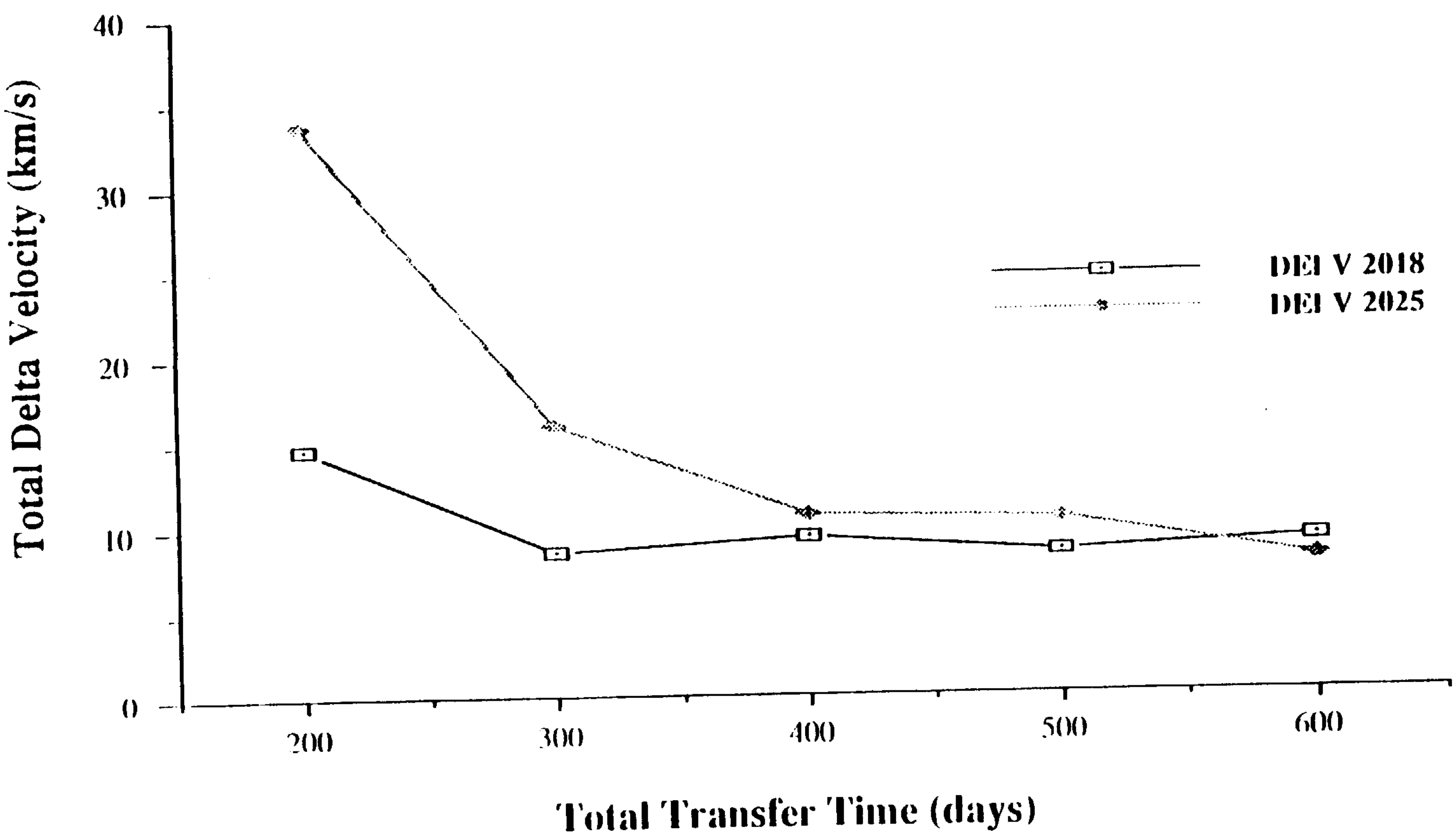
Conjunction Fast Transfer Optimized Data

This set of data were optimized for a fixed totaltransfer times for the 2018 "easy" opportunity and the 2025 "hard" opportunity. This information is applicable to an NTR or Cryogenic All-Propulsive mission. The data shows the same trend as in the data in the preceeding chart all-propulsive curves. If the data is optimized for a fixed transfer time there is still an advantage in going in 2018 for missions with total transfer times less than 550 days. Total transfer times of less than 300 days is possible with an increased price in total ΔV.

Conjunction Fast Transfer Optimized Data

Optimized Trip Time and Stay Time for Fixed Transfer Time

All-Propulsive

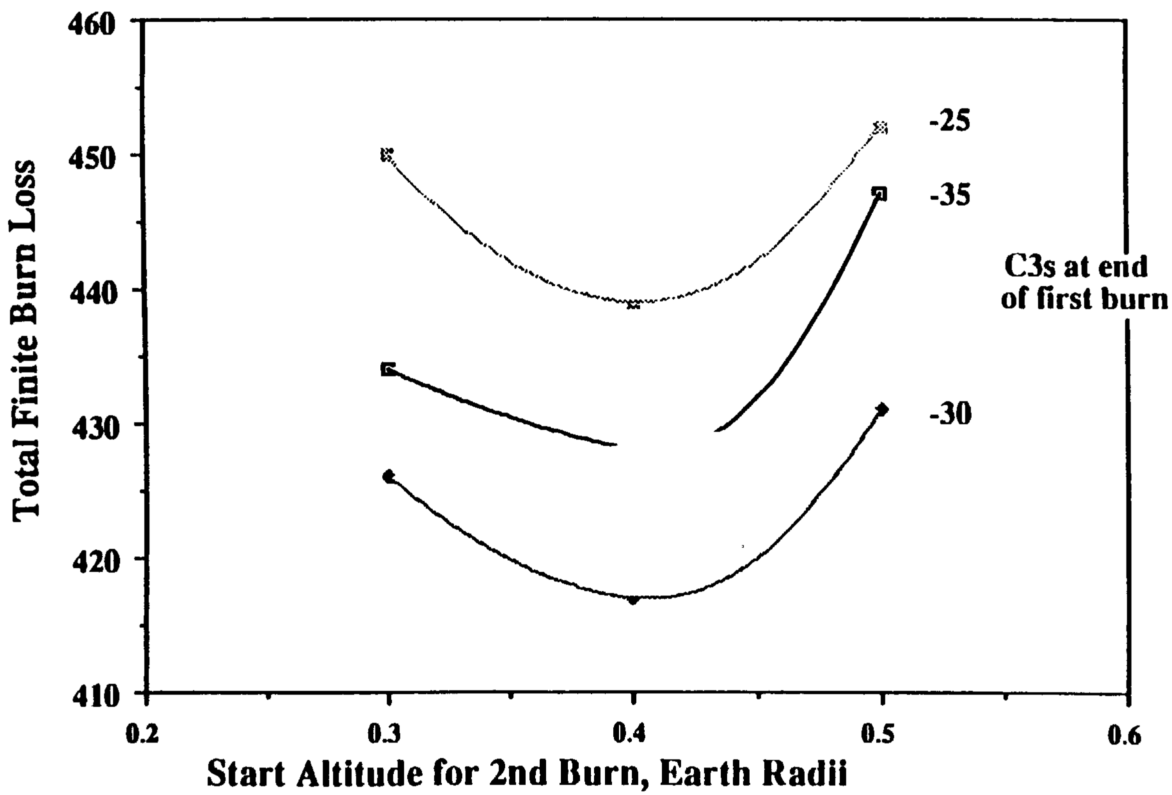


Losses For Two-Burn Trans-Mars Injection

This analysis is a look at the losses in finite burn ΔV for the start altitude required for a second TMI engine burn using the class of engines considered for lunar case main engine use. This shows that it is possible to use these engines to reach the required C3 energy for Mars transit, with more than one TMI burn (broken plane trajectory).

Losses for Two-Burn Trans-Mars Injection

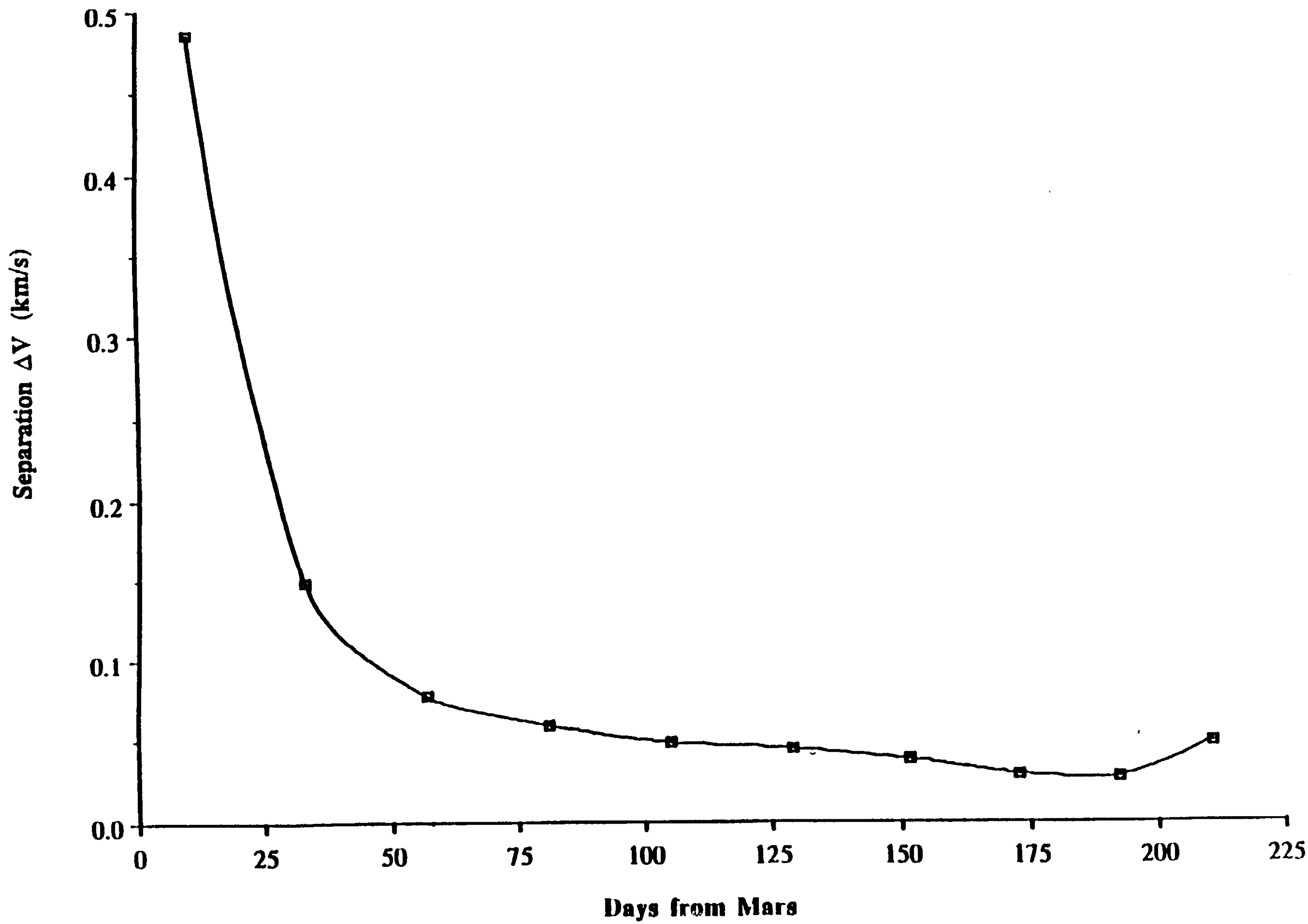
Injection C3 - 20 km2/sec2 T/Wo 0.069 Isp - 481



Separation ΔV vs. Days From Mars for MEV One Day Early Entry Arrival

Before the MTV vehicle arrives at Mars, it must separate with the unmanned MEV, which will aerocapture first, before the manned MTV. The difference between the aerocapture of the unmanned MEV and the manned MTV will be one day. To insure this separation, a separation maneuver must be done before arrival. The cost of this separation in ΔV for is dependent on when it is done. This curve shows that the minimum time from Mars arrival with the minimum required ΔV,(which occurs at the knee of the curve) is 50 days out from Mars. This is for the Level II Reference mission.

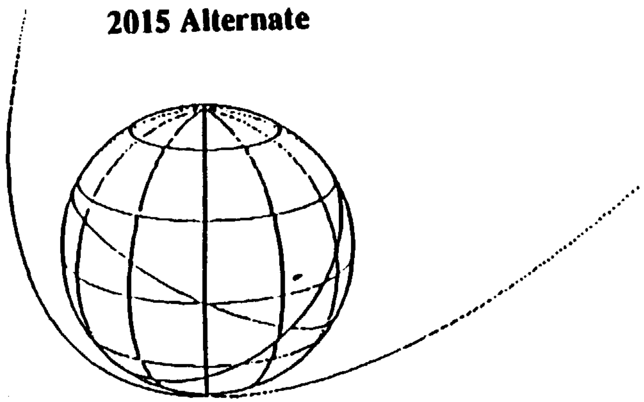
MEV One Day Early Entry Arrival



Capture Trajectory Paths

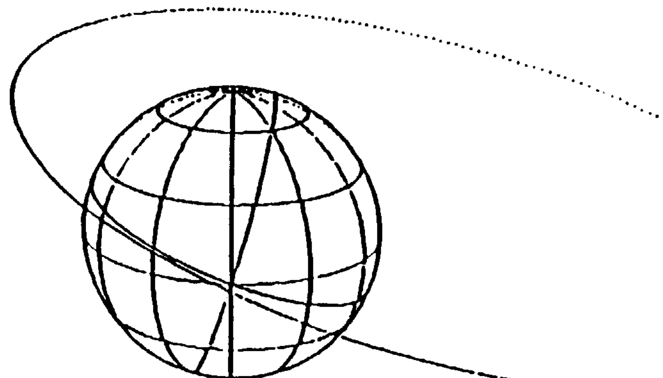
These are pictorial descriptions of the aerocapture arrival inclination for the identified missions. This inclination changes, sometimes drastically, from mission to mission. When the object of the mission is to land at an established base site or a site off the track of aerocapture, either a plane change is required or the need for cross-range capability is establish.

2015 Alternate



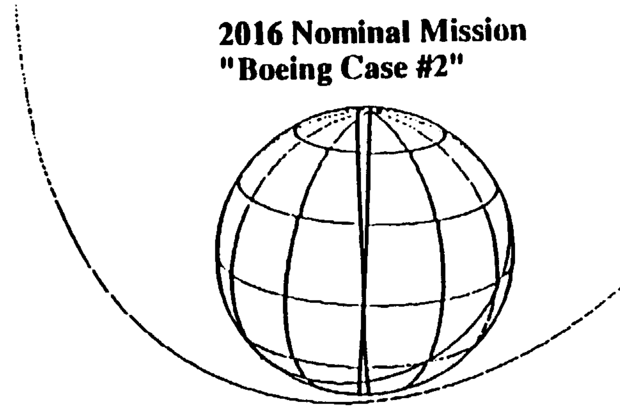
view : latitude 20 deg.
longitude 90 deg.
orbit i = 45.0
periapsis altitude = 250 km

2015 Level II



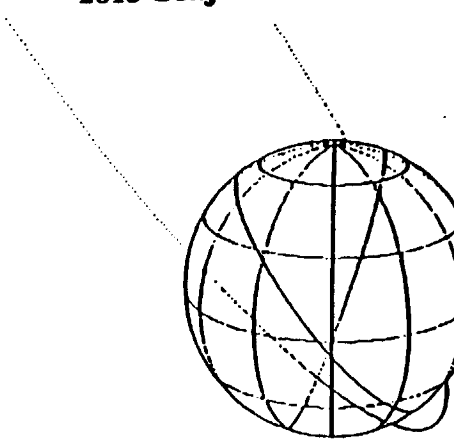
view : latitude 20 deg.
longitude 90 deg.
periapsis altitude = 250 km

2016 Nominal Mission
"Boeing Case #2"



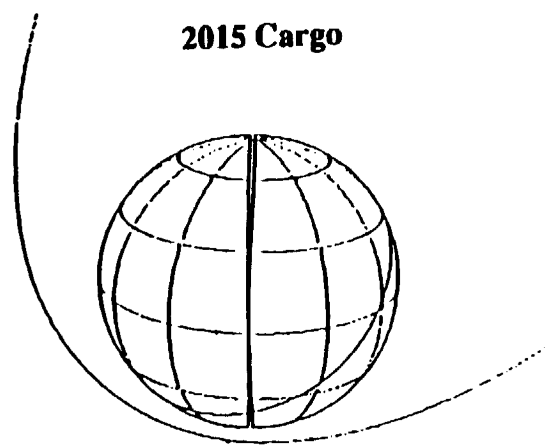
view : latitude 20 deg.
longitude 90 deg.
orbit i = 45 deg.
Periapsis altitude = 500 km

2015 Conjunction Class



view : latitude 20 deg.
longitude 270 deg.
orbit i = 50.0 deg.
periapsis altitude = 500 km

2015 Cargo



view : latitude 20 deg.
longitude 90 deg.
orbit i = 45.0 deg.
periapsis altitude = 500 km

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Performance Parametrics

Note: Contains material formerly in Mission Analysis

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2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period

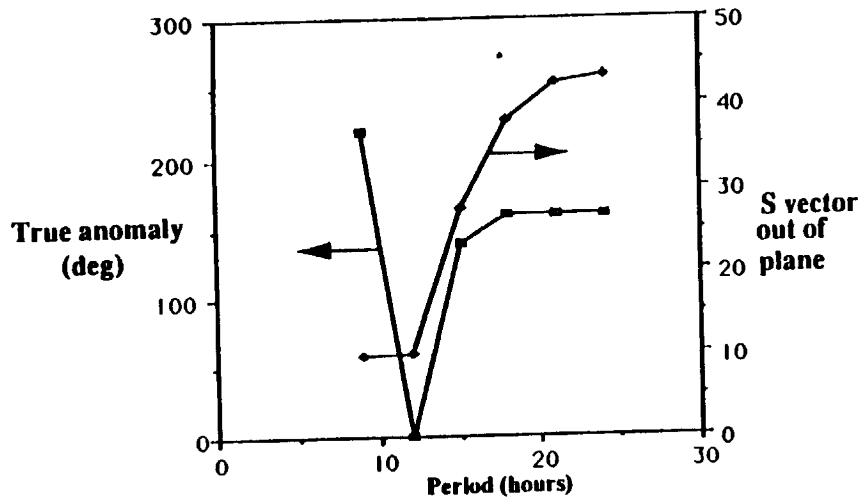
Minimum departure delta V occurs when the S vector is in plane and departure true anomaly is close to periapsis.

./STCAEM/mlc/12June90

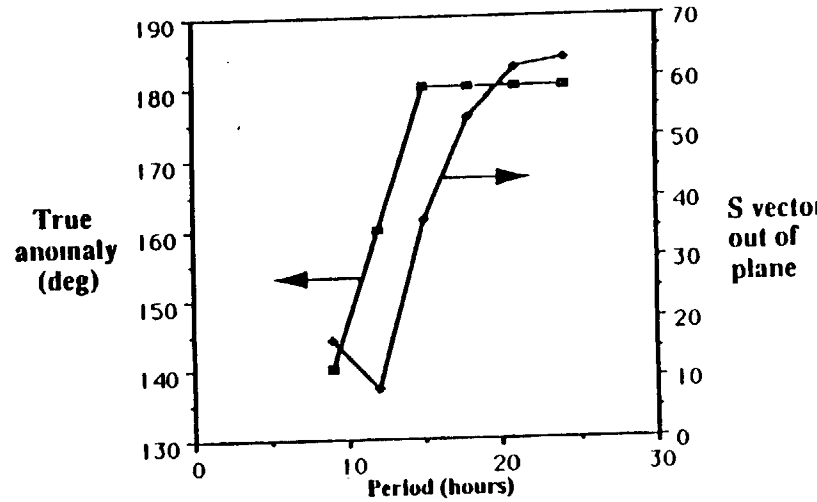
2010 Conjunction
S Vector and True Anomaly vs Parking Orbit Period

ADVANCED CIVIL SPACE SYSTEMS — BOEING

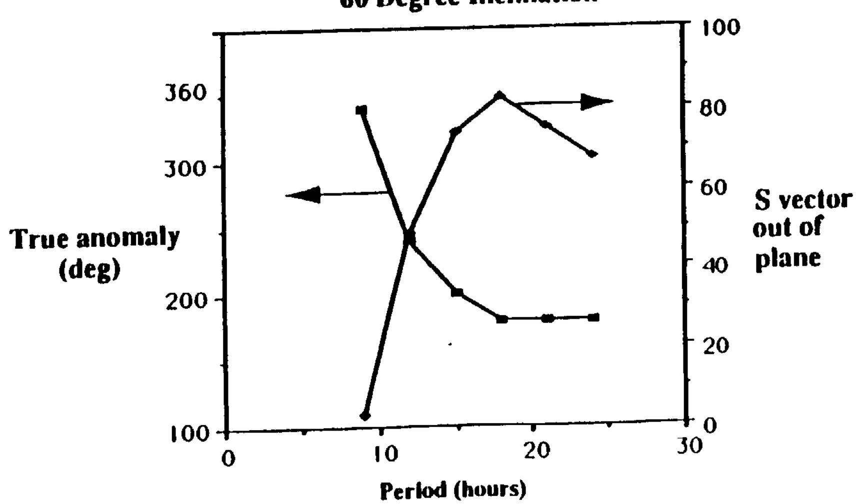
20 Degree Inclination



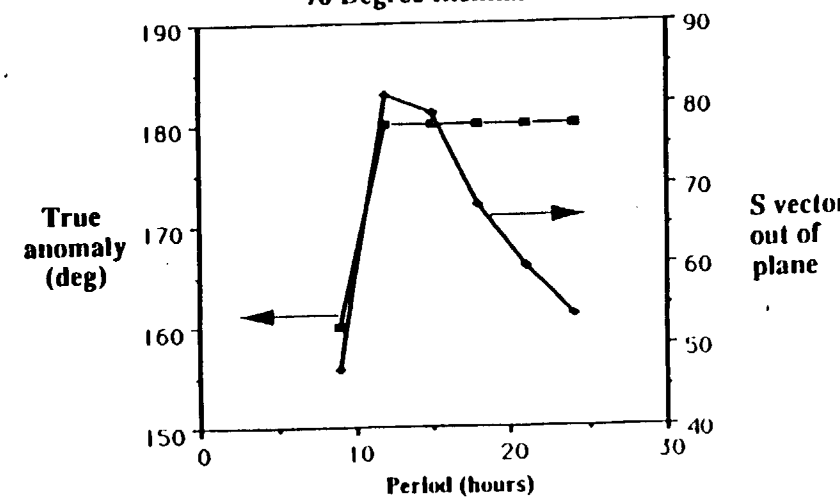
40 Degree Inclination



60 Degree Inclination



70 Degree Inclination



/STCAEM/grw/31May90

2010 Conjunction Mission
Orbital Parameters vs Parking Orbit Period

Minimum Mars departure ΔV of approximately 1.2 km/sec occurs for parking orbits with inclinations of 30° and 60° and period of 9 hours each.

Periapsis lighting angle is adequate for the parking orbits with inclinations of 30° and 60° and period of 9 hours each.

Minimum S vector out-of-plane occurs when the true anomaly at departure is closest to periapsis and the orbital period is 9 hours.

Periapsis latitude for a 30°, 9 hour parking orbit provides a landing coverage for landing sites between 38° to 50° north latitude; periapsis latitude for 60°, 9 hour parking orbit provides landing site access to landing sites between 5° south to greater than 20° north latitude.

STCAEM/mk/12June90

3

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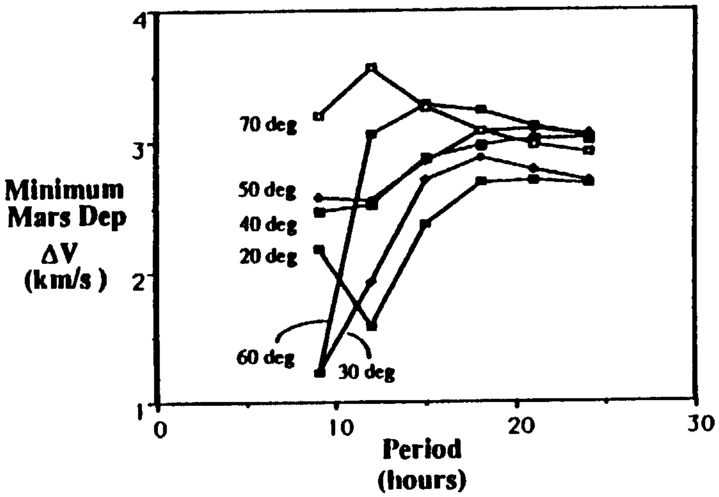
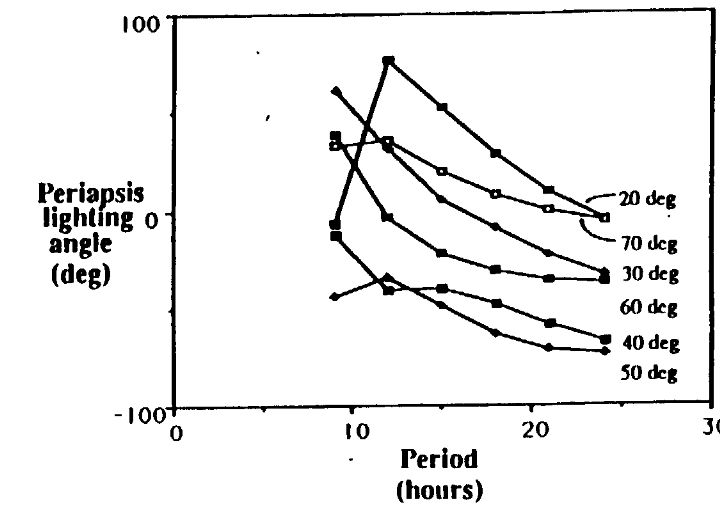
2010 Conjunction
Orbital Parameters vs Parking Orbit Period

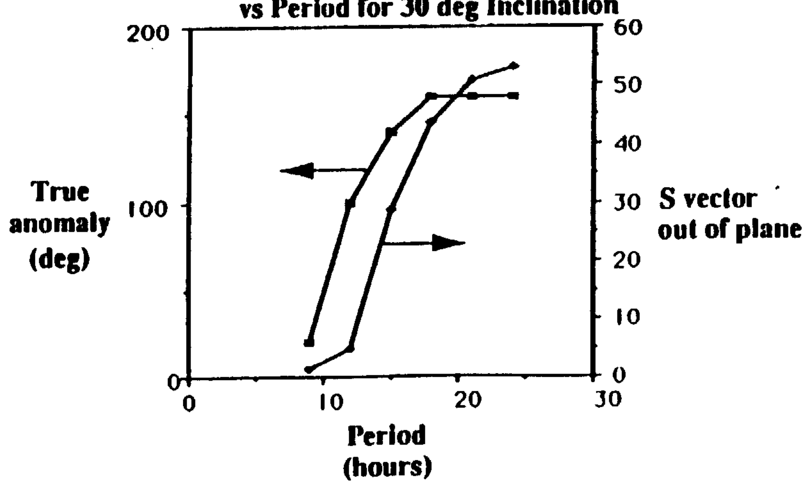
Minimum Mars Departure ΔV vs. Period



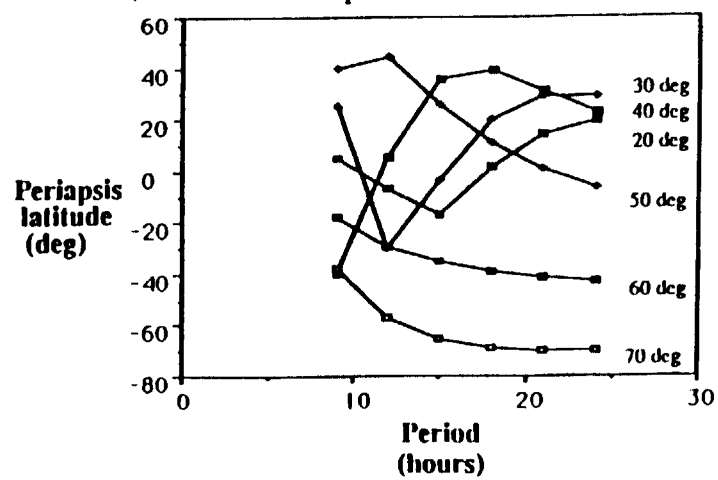
Periapsis Lighting Angle vs Period



S Vector Out of Plane and True Anomaly vs Period for 30 deg Inclination



Periapsis Latitude vs Period



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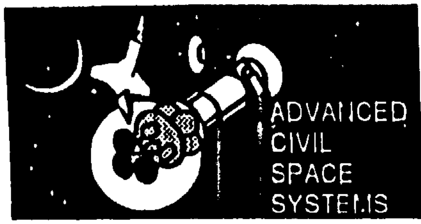


Aerocapture GN&C Analyses

Analysis Type		Results Obtained
• Closed-form zero lift approximation; fixed exponential atmosphere	√	Depth of penetration versus ballistic coefficient and entry velocity
• Fixed-lift integrated trajectories; 2-DOF; fixed tabulated atmospheres	√	Corridor height and g level vs. available L/D and entry velocities; entry conditions
• Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; fixed tabulated atmosphere	√	Trajectory designs for aerocapture, considering vehicle lift modulation capability and rates
• Modulated lift integrated trajectories; 3-DOF or 3-1/2 DOF; variable atmosphere	√	Development of guidance schemes and laws; assessment of errors induced by atmoshpere unpredictability
• 6-DOF integrated trajectories with simulation of vehicle flight control system; variable atmosphere		Accurate assessment of vehicle capabilities for aerocapture; detailed design requirements for aerobrakes and flight control systems

Guidance, Navigation & Control

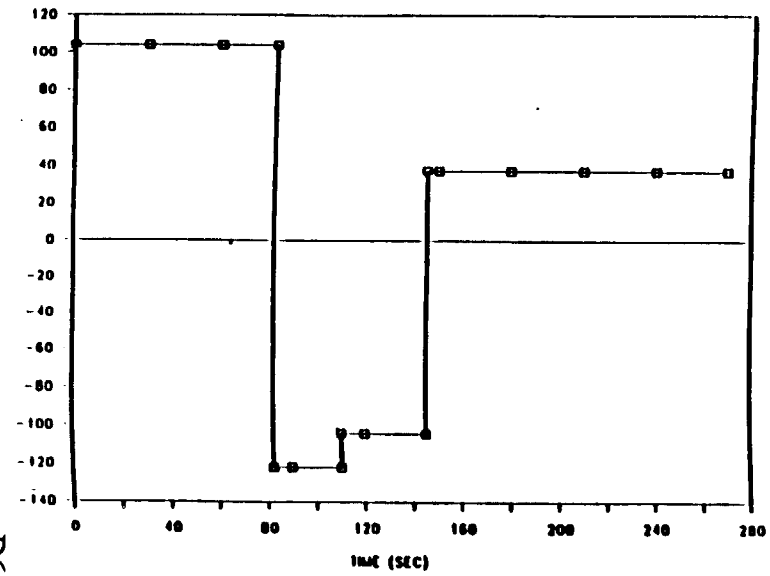
This shows another example of the trajectory design, using the OPTIC code. We are using two different GN&C codes to cross-check results. OPTIC, developed by Boeing-Seattle, optimizes with constraints. The other, AEROPASS, developed by Boeing-Huntsville, optimizes switch points and exercizes guidance laws. Constraints must be represented by penalty functions with this routine.

ADVANCED CIVIL SPACE SYSTEMS

Guidance Navigation & Control

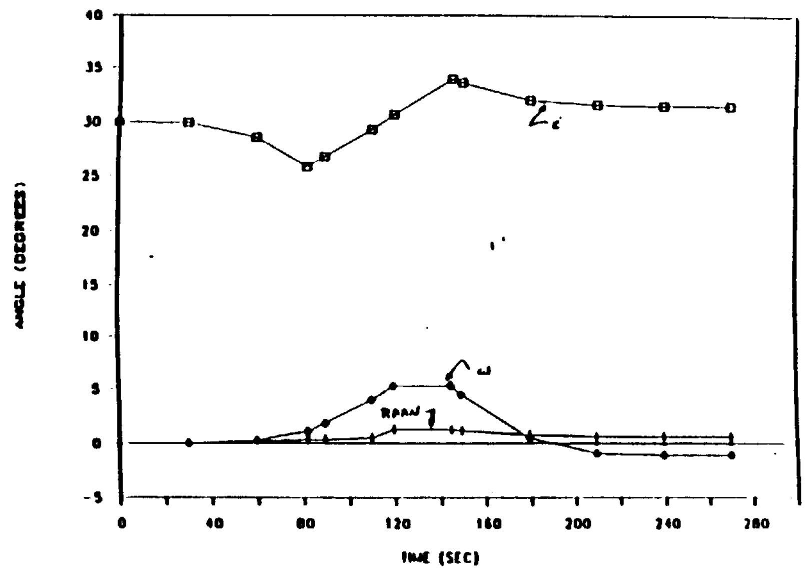
BOEING

LIFT MODULATION VIA BANK ANGLE MOD.

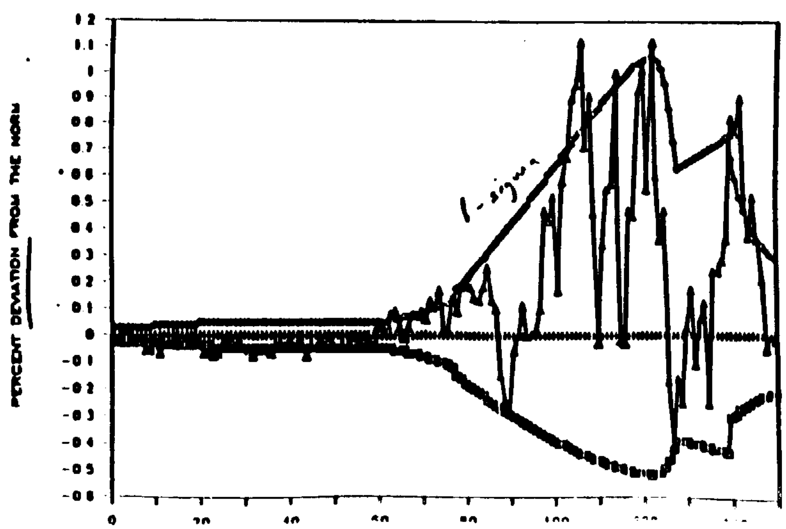


< Guidance control through the Mars atmosphere is done by controlling bank angle in a "slalom course" motion

INC,RAAN, & ARGPER PROFILES



This is the resultant inclination, right ascension angle and argument of periapsis (position with respect to the planet) for the "slalom course" run >



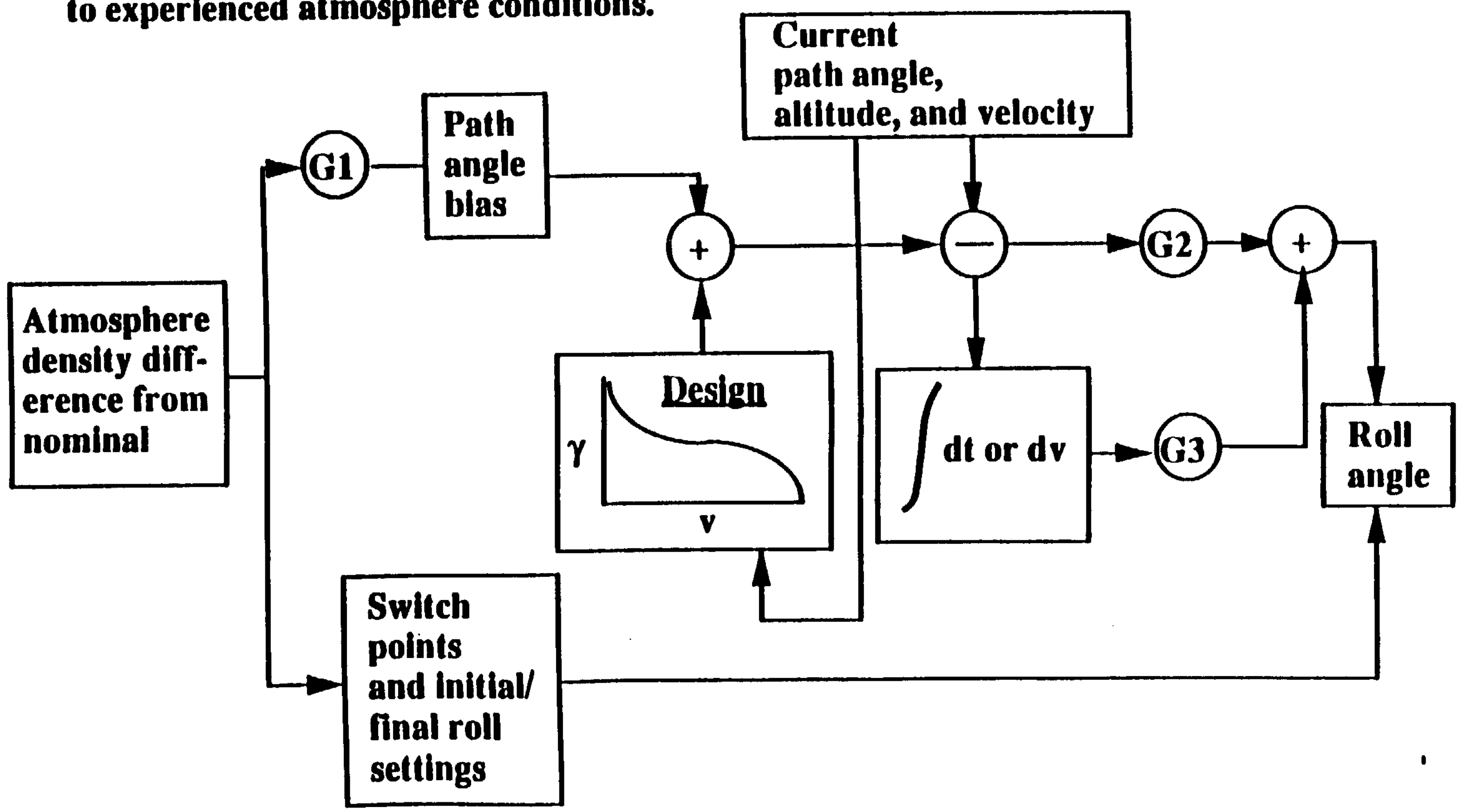
< comparison of the atmospheric deviation obtained with the MARSGRAM and Optic codes

Guidance Schemes for Aerocapture

The facing page identifies the requirements for aerocapture GN&C, summarizes two approaches to adaptive guidance, and illustrates a gain-scheduling scheme presently under investigation. We also plan to investigate the real-time re-optimization scheme.

Guidance Schemes for Aerocapture

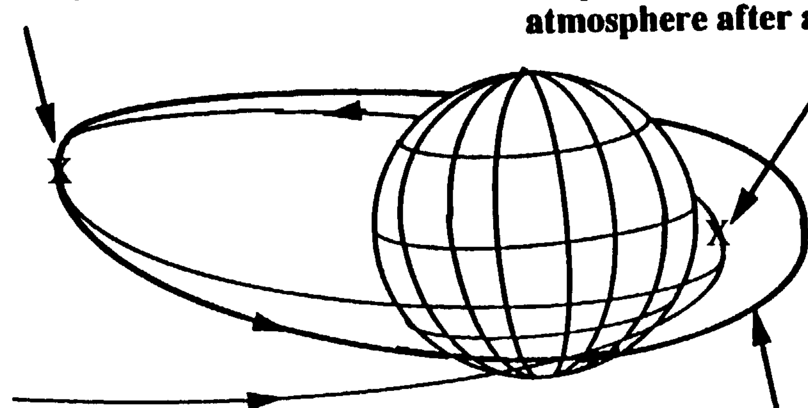
- Requirements - minimize changes in inclination and line of nodes. Attain desired line of apsides and apoapsis altitude.
- Maximum performance - redesign trajectory optimization and constraints every few seconds, but requires very high computer performance.
- Less performance but may be adequate - use an adaptive gain scheduling scheme that adjusts to experienced atmosphere conditions.

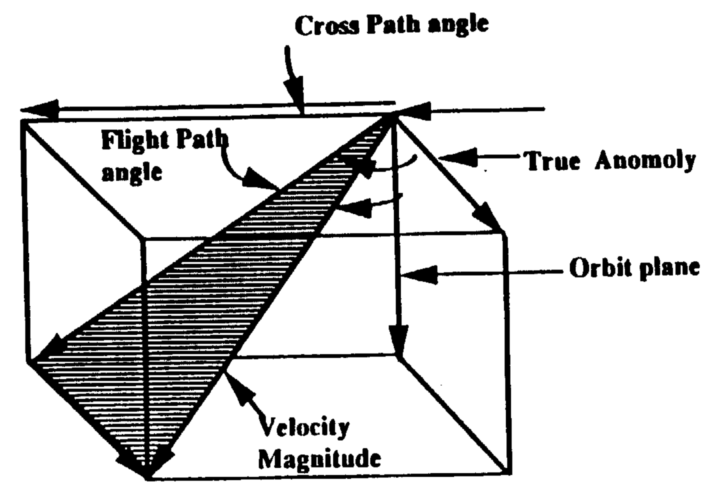


Orbit Correction Analysis

The next three pages illustrate a method of correcting exit conditions with two burns, and show some preliminary results of calculations of the delta V required for each burn as a function of the magnitude of exit errors.

Orbit Correction Analysis





- Burn at V1 will - align the apsides
  - set the apoapsis height to the desired final orbit height
- Burn at V2 will - raise periapsis height
  - correct inclination errors

- State vector errors are:
  - flightpath angle
  - crosspath angle
  - velocity magnitude
- Position Vectors errors are:
  - true anomaly
  - orbit plane

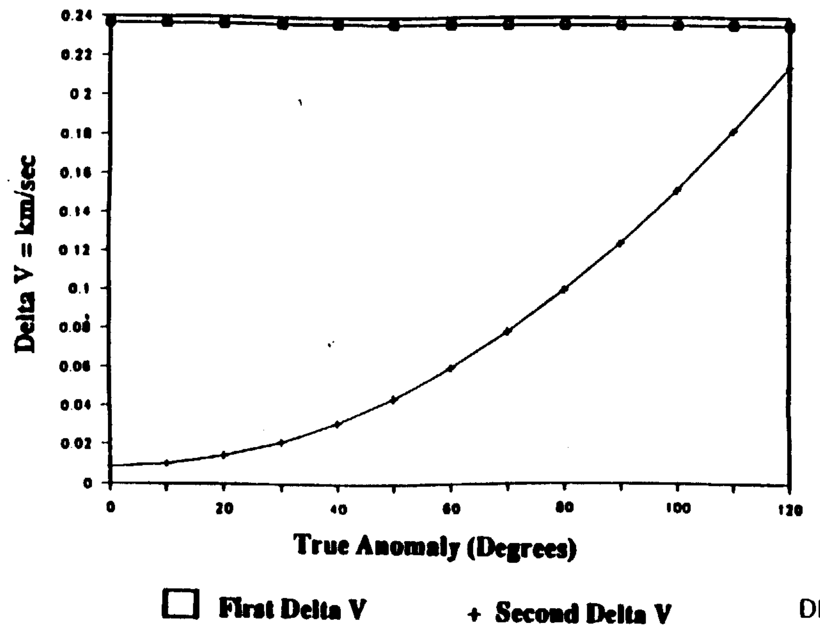
Effect of Burn Location on ΔV Penalties

Page 1

BOEING

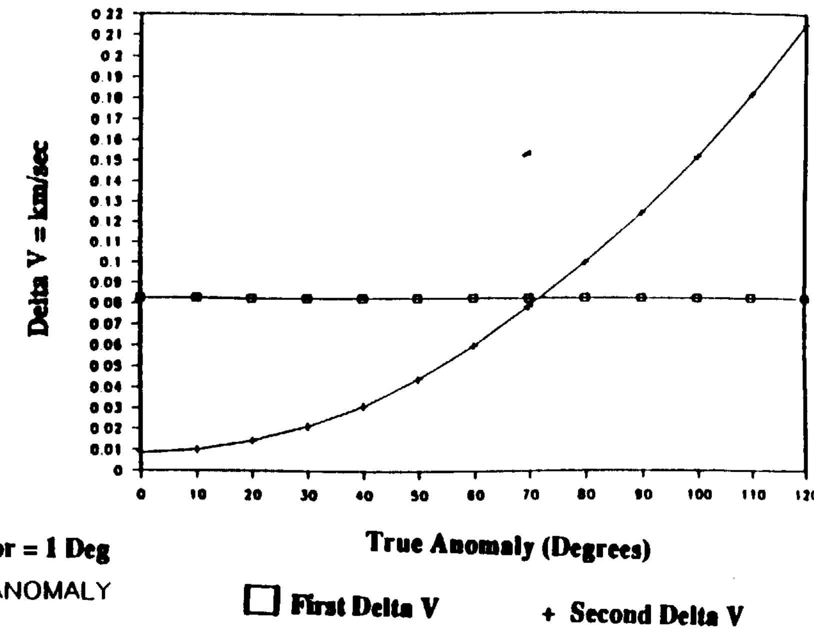
Velocity Magnitude Error = 0.95

DELTA V VS. TRUE ANOMALY



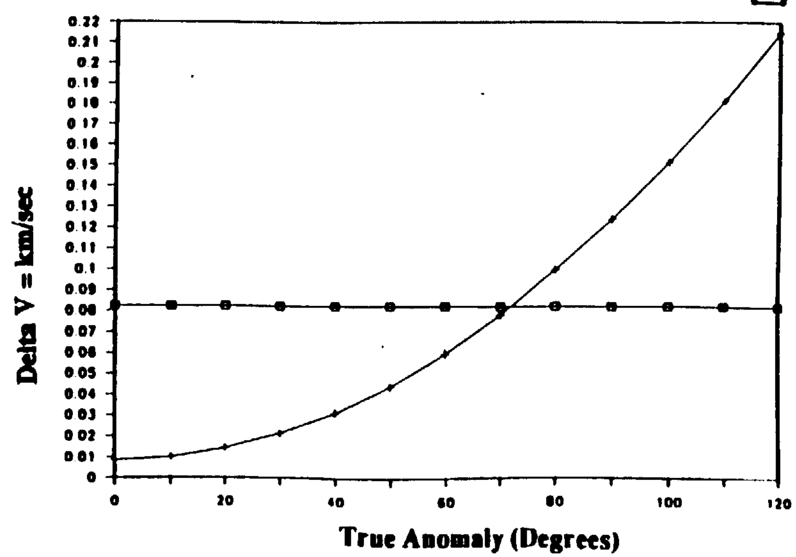
Flight Path Angle Error = 1 Deg

DELTA V VS. TRUE ANOMALY



Cross Path Angle Error = 1 Deg

DELTA V VS. TRUE ANOMALY

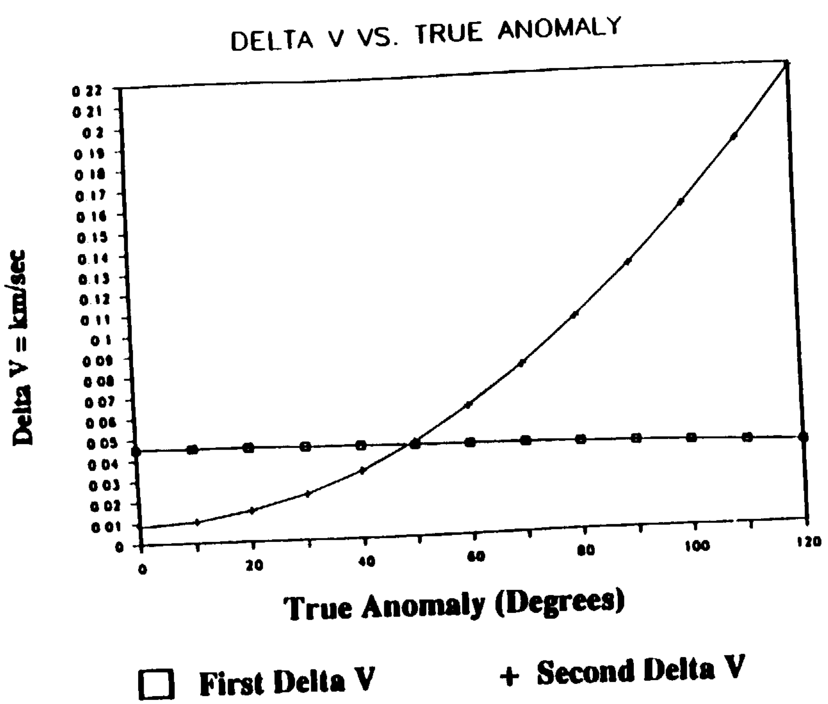


Effect of Burn Location on ΔV Penalties

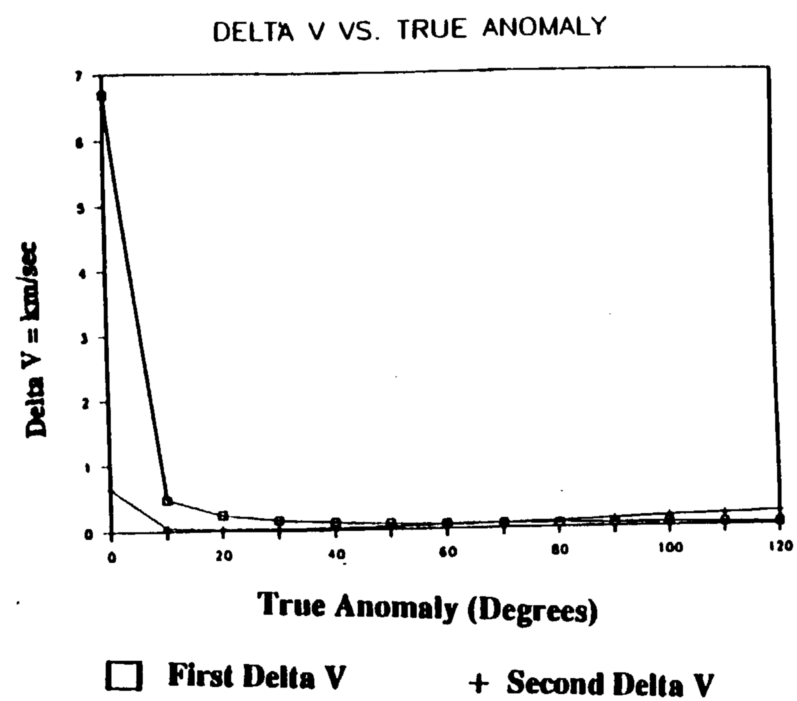
Page 2

BOEING

True Anomaly Error = 1 Deg

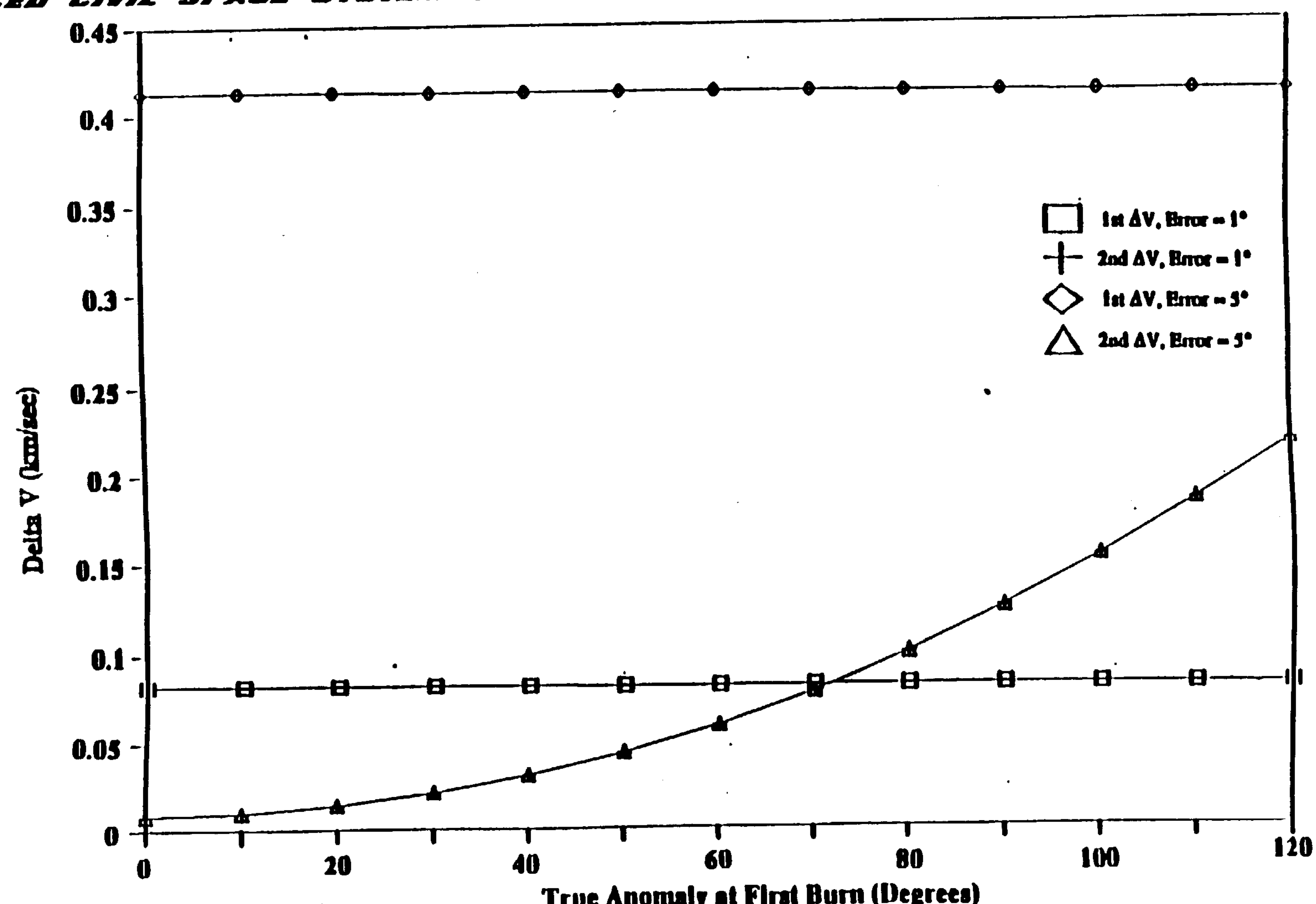


Position Plane Error = 1 Deg



The delta velocity is independent on the true anomaly at first burn but is dependent on the flight path angle error. The two flight path angle errors examined were one degree and five degrees. Even though the delta V's for the different flight path angles are significantly different, the delta V for the second burn is not dependant on the error in the first burn. The total correction delta V is minimized by performing the first burn close to the periapsis where the true anomaly is zero.

Flight Pa 1 Angle Error on Delta V



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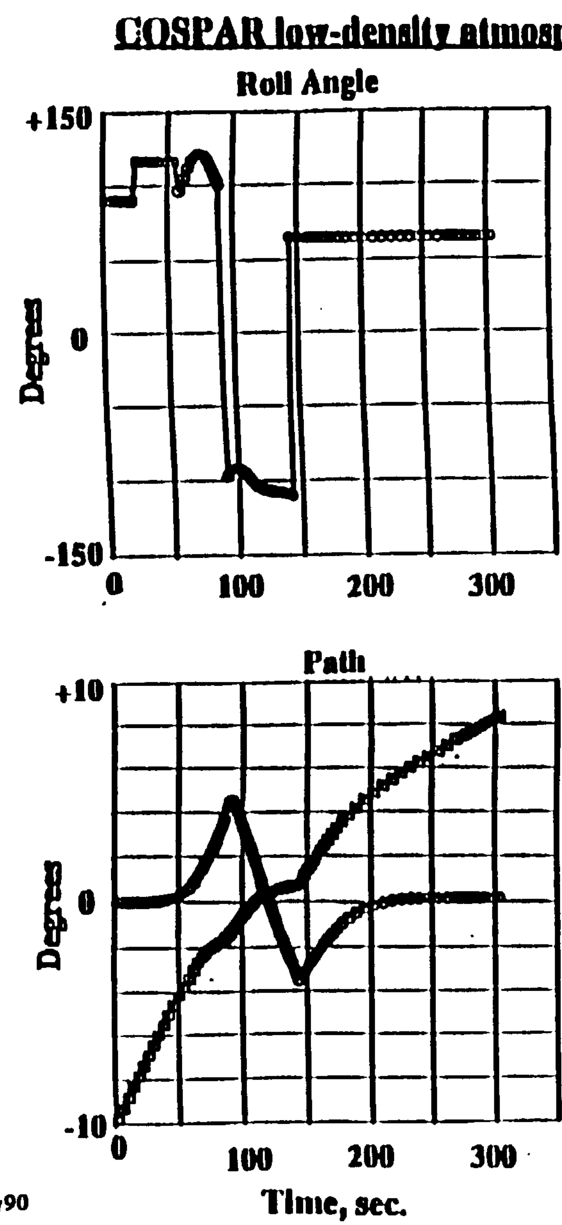
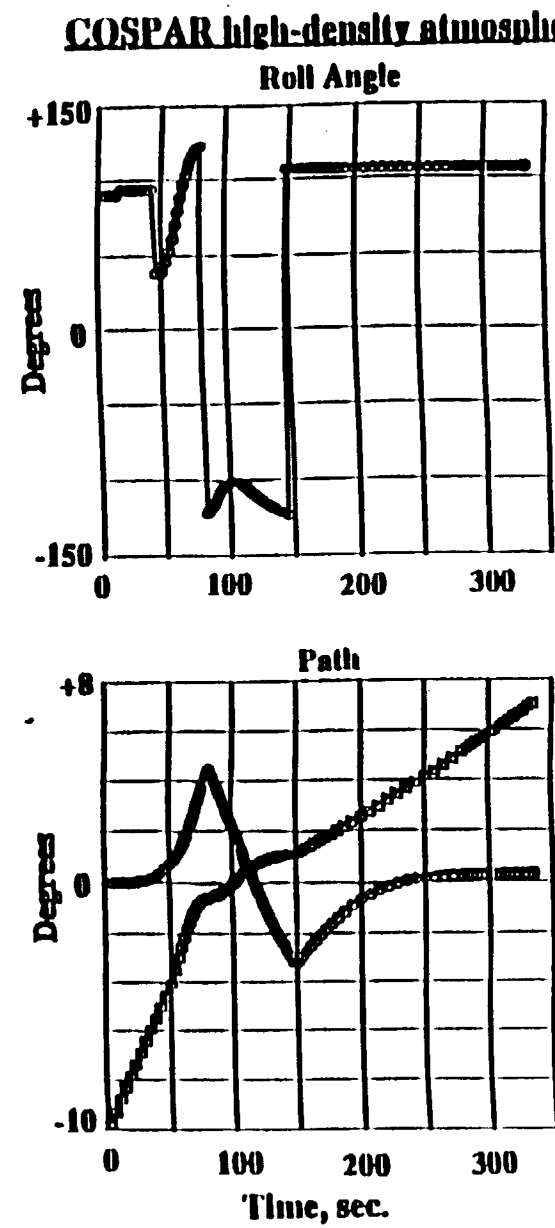
In the early part of the trajectory, the roll angle is set by the gravity loading which is an indication of the type of atmosphere encountered. In the early to mid portion of the trajectory, the roll angle is used to minimize the flight path angle error which is based on the difference between the actual flight path angle and the predetermined optimum flight path angle. In the latter part of the trajectory the roll angle is used to minimize the line of apsides error. Examples of guided trajectory profiles are provided for low and high density atmospheres.

Guided Trajectory Examples





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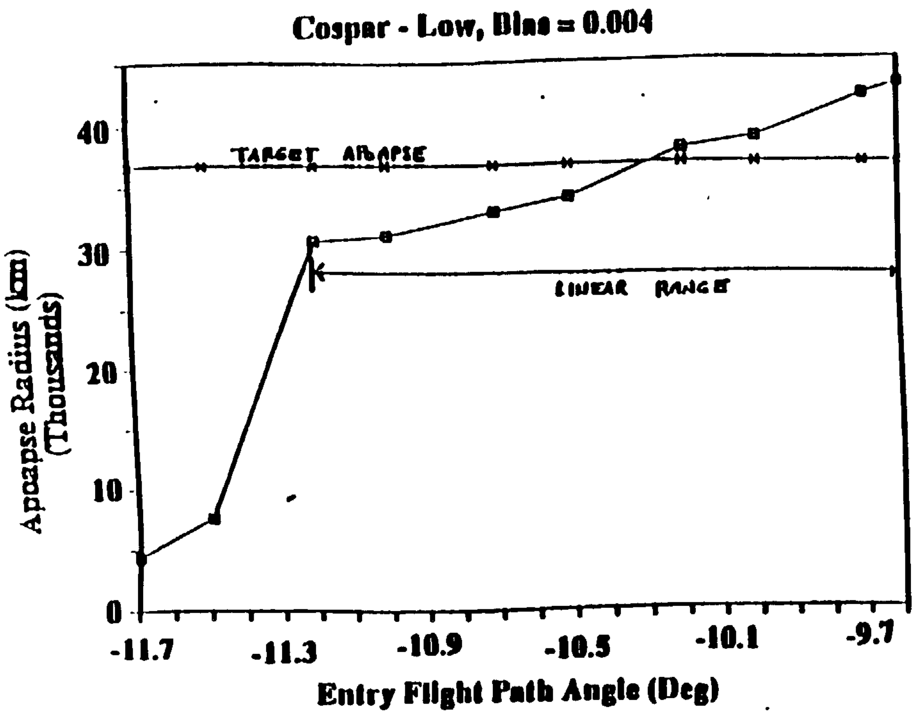
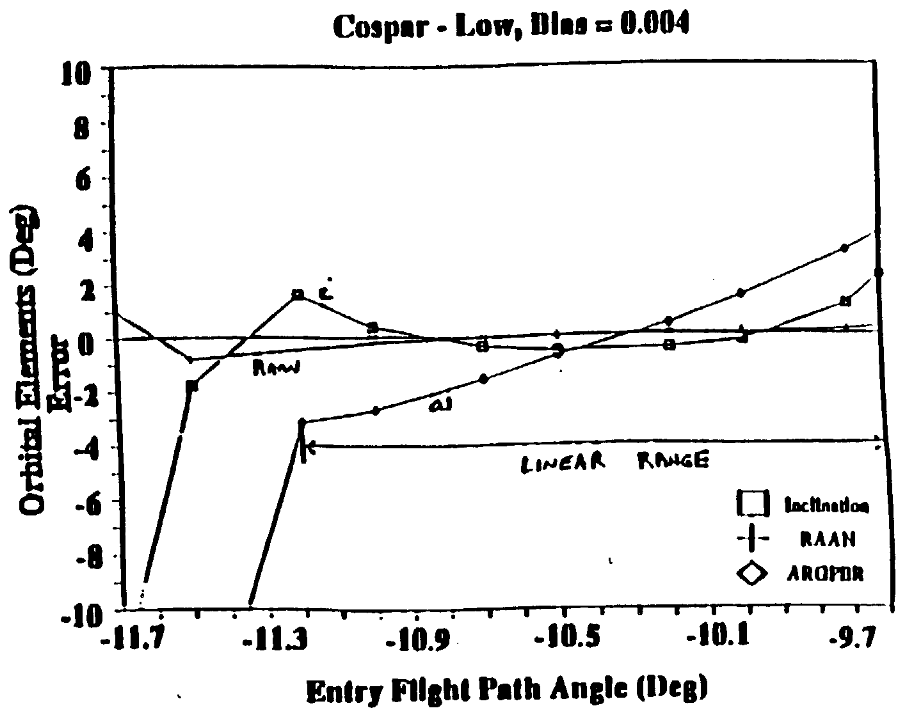
For the hyperboloid vehicle it is illustrated that the constraint can be met with an entry flight path angle of approximately ± 0.75°. This is illustrated for the COSPAR low atmosphere.

Parking Orbit Constraint Errors





Mars Aerocapture Trajectory - Finite Roll Rate

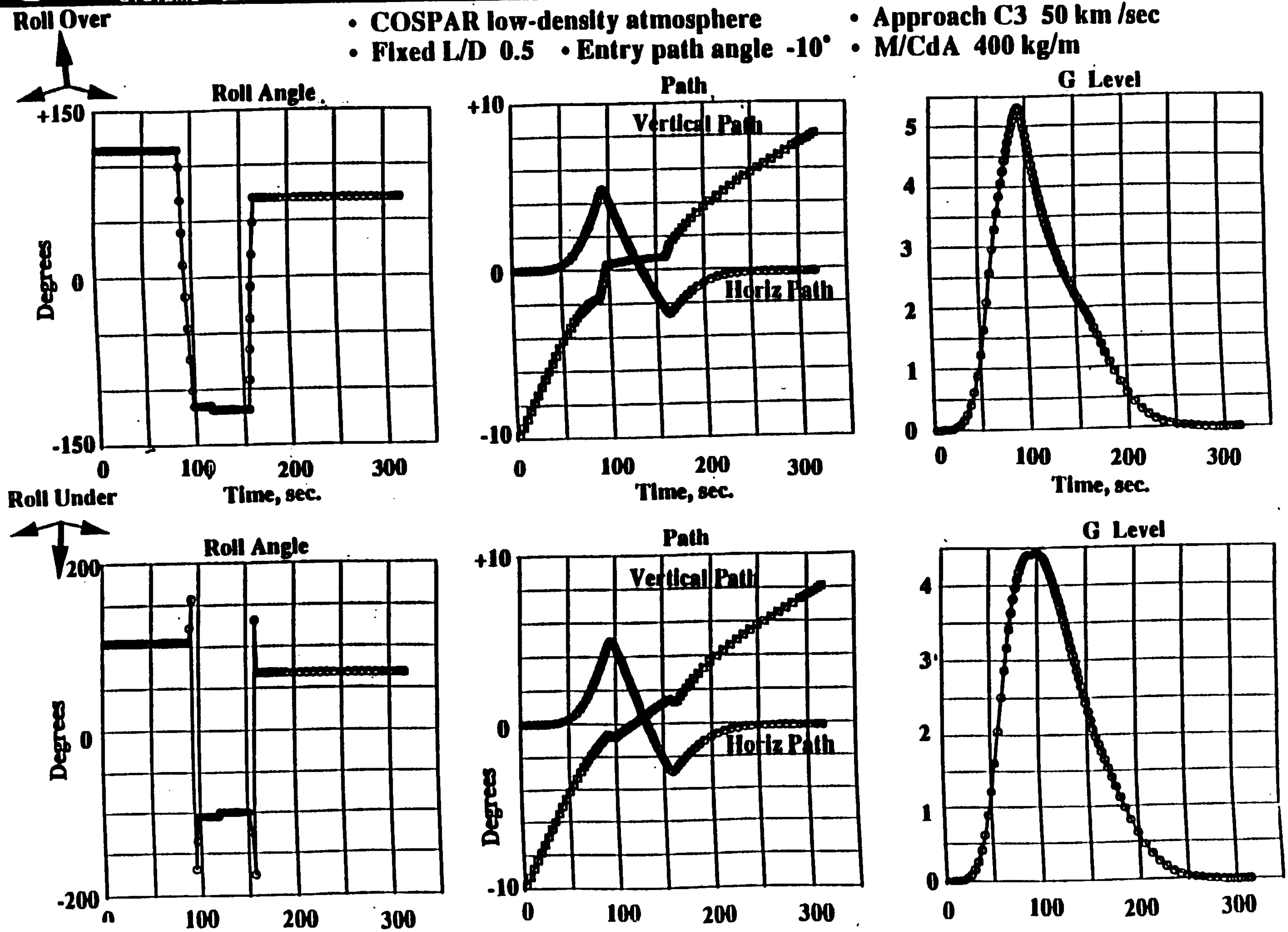
This figure shows the effects of finite roll rates on the trajectory design. In going from left to right and back, the lift vector may be rolled over the top, or under. Because this perturbs the vertical path, the rest of the trajectory design must compensate by going a little deeper (roll over) or a little less deep (roll under). These results show that the effect on the vertical path is less with the roll under, and that the maximum deceleration is less, leading to a clear preference for "roll under".

Mars Aerocapture Trajectory - Finite Roll Rate

ADVANCED CIVIL SPACE SYSTEMS

BOEING

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km /sec
- M/CdA 400 kg/m



Mars Aerocapture Trajectory Design Approach

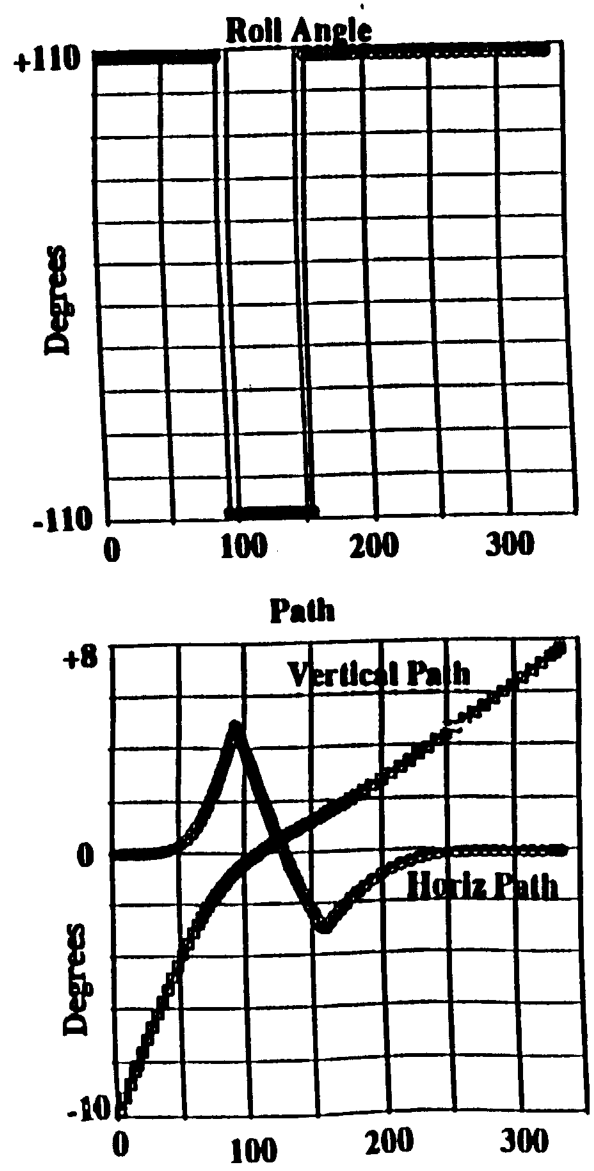
The trajectory design approach is tailored to a roll-only control scheme. Excess lift is dissipated by veering the trajectory to the left and to the right in a dog-leg or "slalom" maneuver. The illustration on the left shows a nominal symmetric design, with roll angles of 107° (0° is straight up). This applies a net vertical lift coefficient of about -0.15. The asymmetric design on the right enables control of the line of apsides, so that the range of atmospheres represented by the COSPAR low and high density atmospheres can be navigated, from the same entry conditions, to realize the same capture orbit, within reasonable delta V budget for post-exit correction. For the low-density atmosphere, the roll angle is greater during penetration than during exit. For the high-density atmosphere, the reverse is true.

Mars Aerocapture Trajectory Design Approach

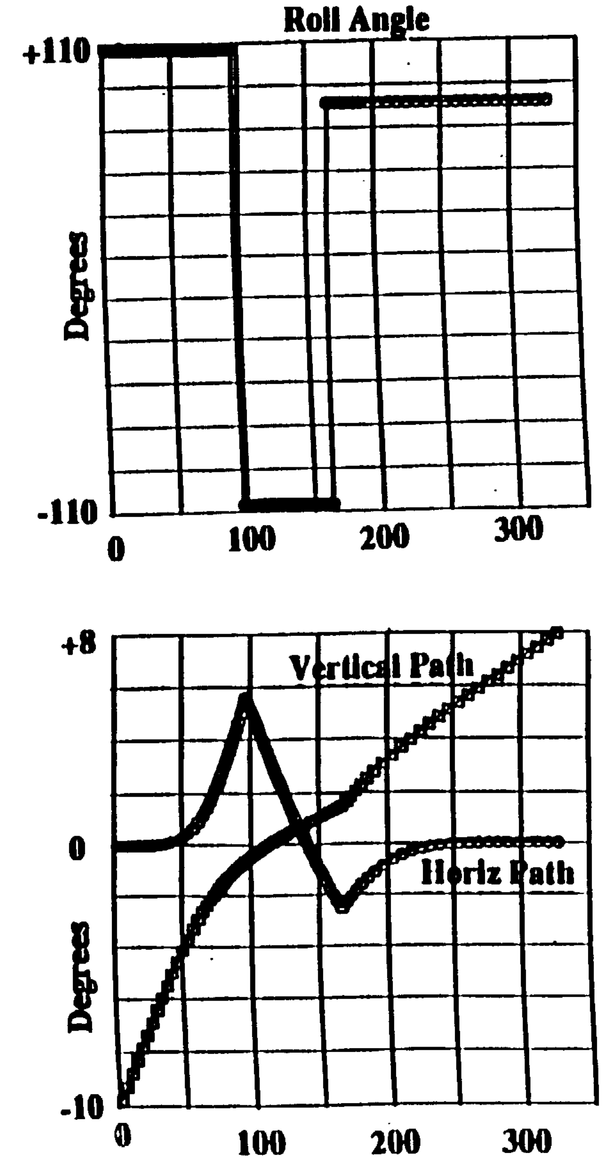
ADVANCED CIVIL SPACE SYSTEMS

BOEING

- COSPAR low-density atmosphere
- Fixed L/D 0.5
- Entry path angle -10°
- Approach C3 50 km /sec
- M/CdA 400 kg/m



Basic design is symmetric



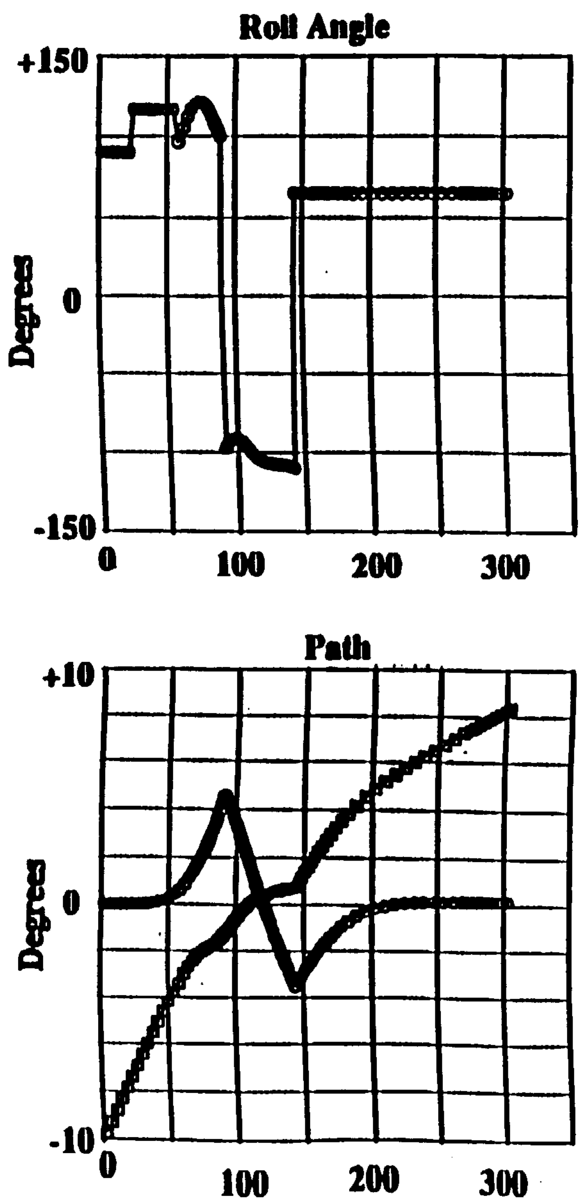
Asymmetric roll program enables line-of-apsides control

Mars Aerocapture - Guided Trajectory Examples

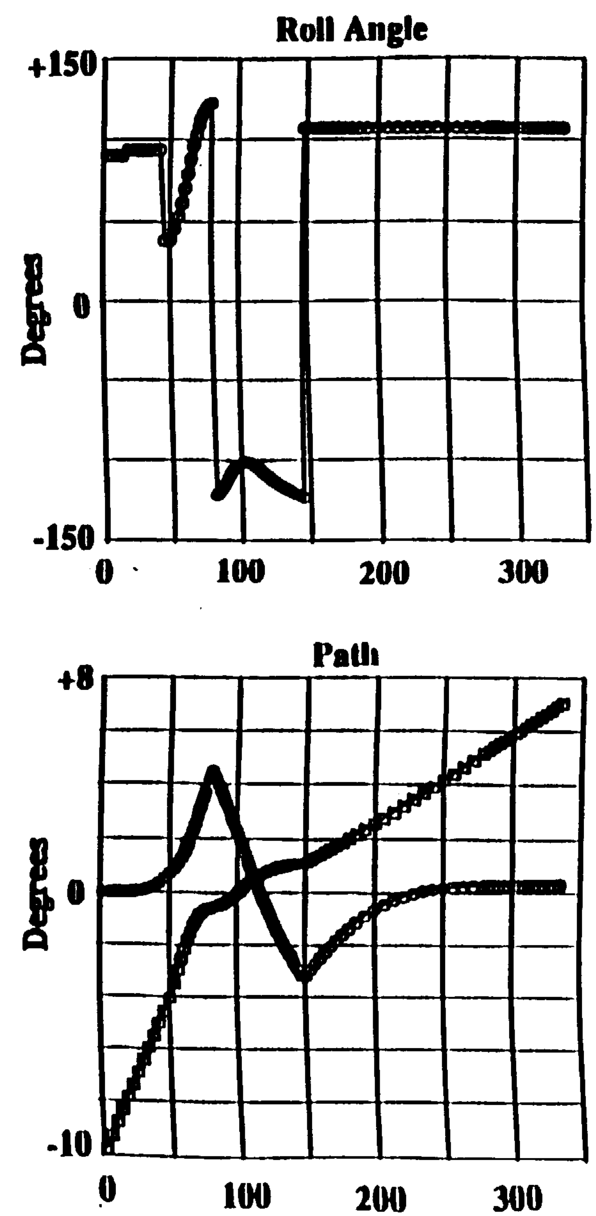
This shows preliminary results of trying the gain-scheduling scheme. Gains have not been optimized. The same entry conditions were used with the high and low-density atmospheres. The guidance scheme gave good results on all exit conditions except line of apsides, and fair results for that parameter.

Mars Aerocapture - Guided Trajectory Examples

COSPAR low-density atmosphere



COSPAR high-density atmosphere

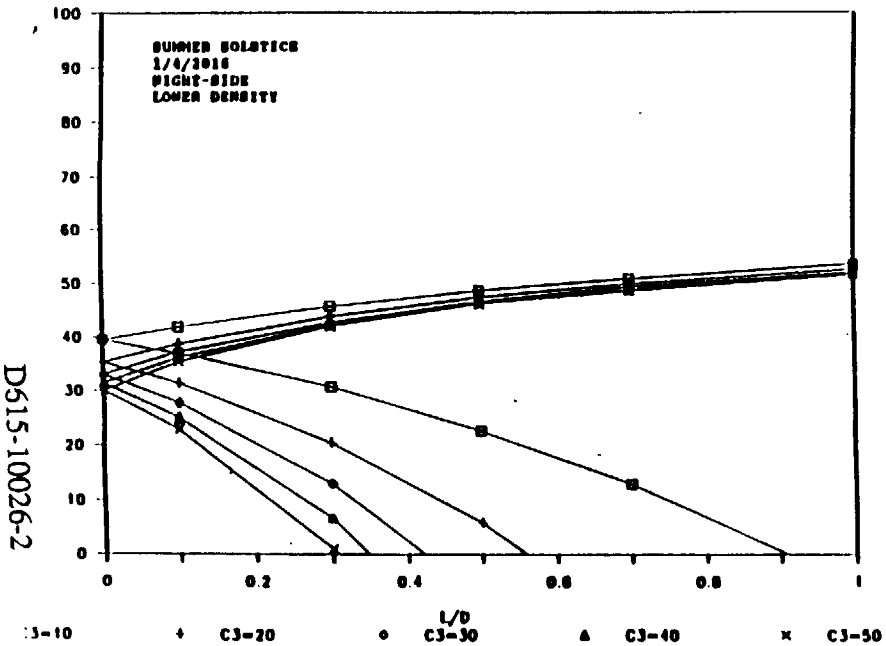


"Slalom Course" Maneuver Profiles

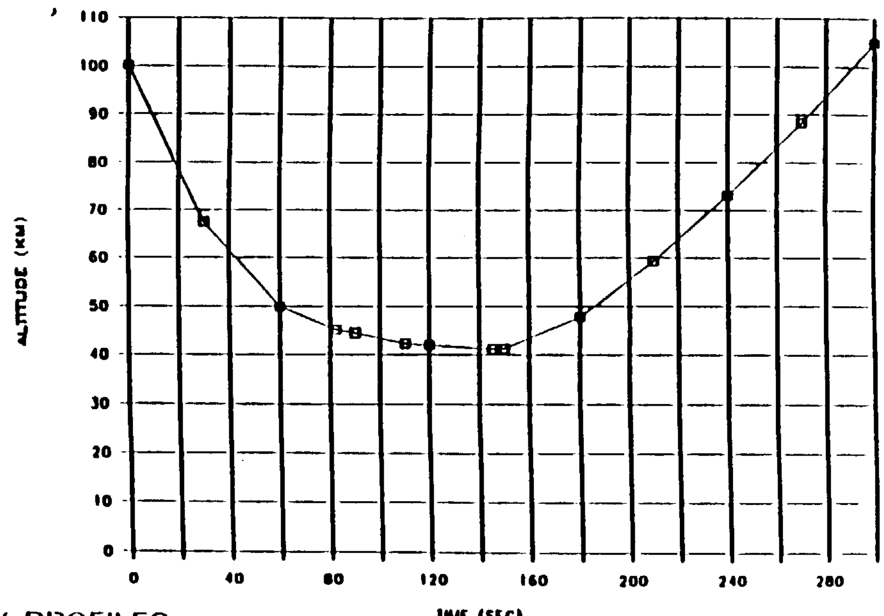
This simulation, with the OPTIC code, examined the effects on a trajectory design of typical atmosphere density variations predicted by MARS-GRAM. The most significant effects were a significant reduction in exit velocity and a large rotation of the line of apsides. No adaptive guidance was simulated. The result shows a clear need for adaptive guidance.

Atmospheric Entry Conditions for "Slalom Course" Maneuver

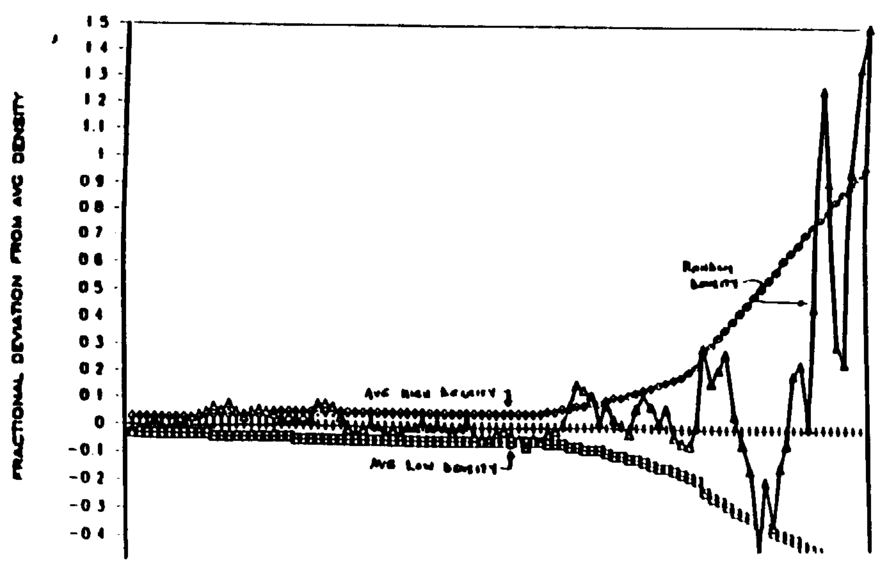
CORRIDOR PROFILE



TRAJECTORY PROFILE



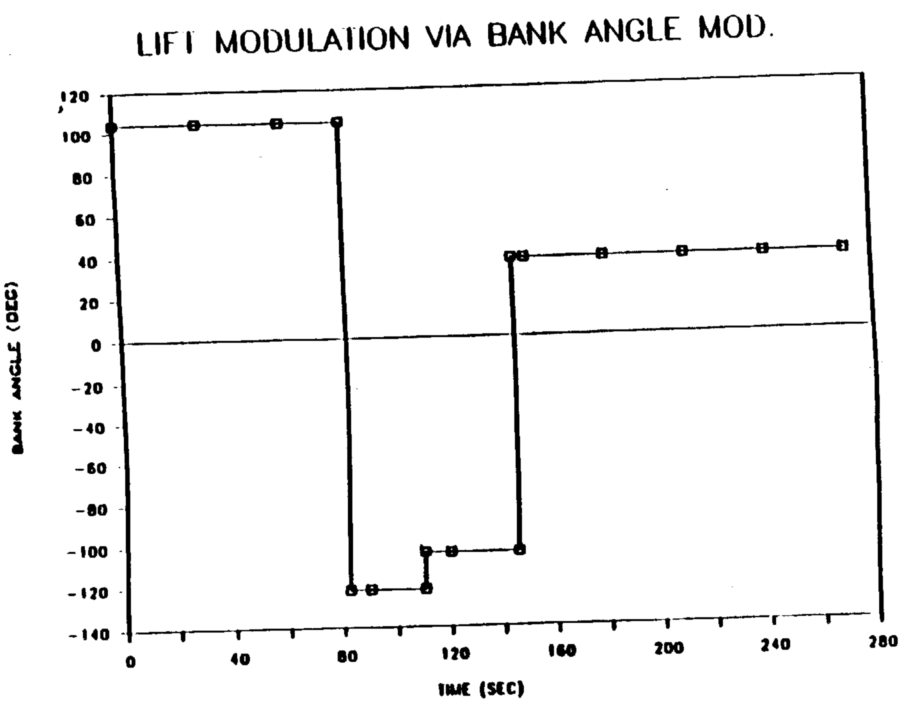
DENSITY PROFILES

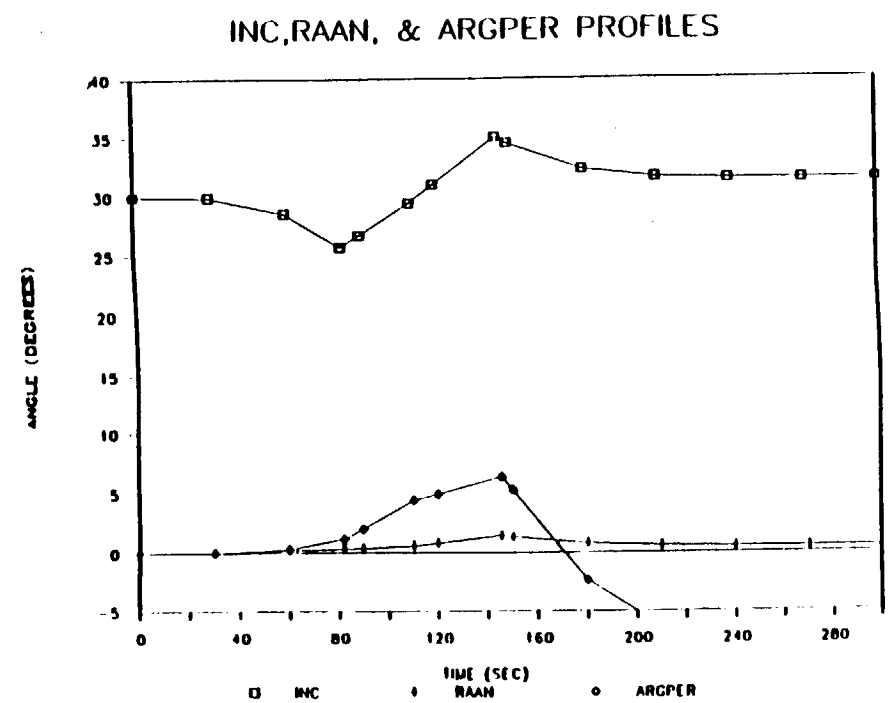


Atmosphere Entry Conditions for "Slalom Course" Maneuver

An additional display of the trajectory design is shown here. Corridor height parametrics are on the left. A typical trajectory profile for a relatively dense MARS-GRAM atmosphere is on the upper right. Typical MARS_GRAM atmosphere desnity predictions are shown on the lower right.

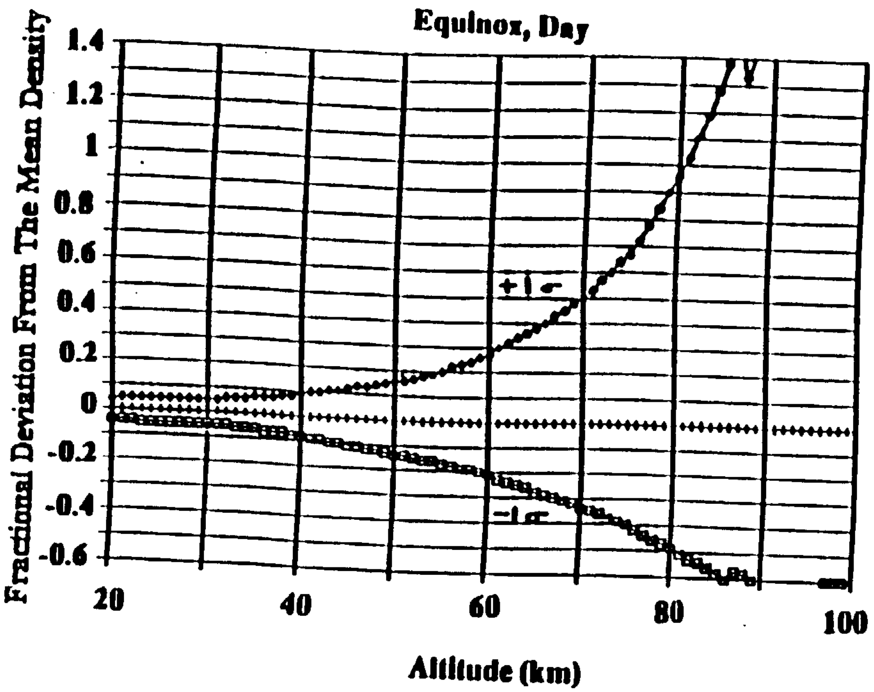
"Slalom Course" Maneuver Profiles

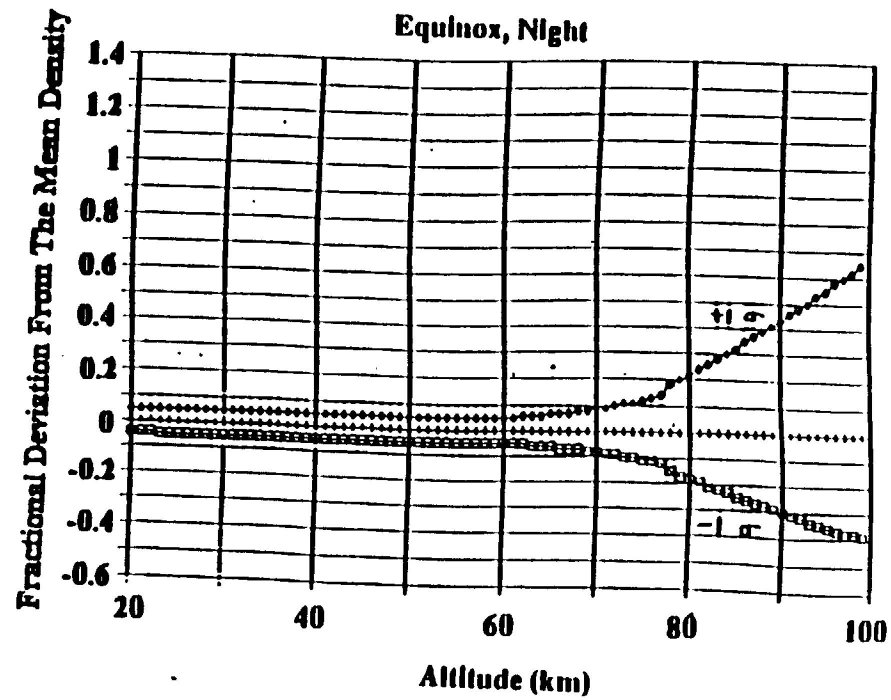




According to MarsGRAM, the one sigma density variation about the mean is illustrated for day and night at the equator during spring equinox of the year 2016. The wider envelope for the equinox day profile is due to the diurnal bulge.

Variations of Density Envelope





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A set of synthetic-density wave equations are given on the following These wave equations were used to determine aerocapture guidance sensitivities from density variations (worst case) that may be encountered during Mars Aerocapture.

Synthetic Density Wave Equations

- Horizontal sine-wave density scaling

  DENS = DENS*[1.0 + 0.3*SIN((Y/W)*2Pi)]

- Horizontal sine-wave density and vertical density ratio scaling

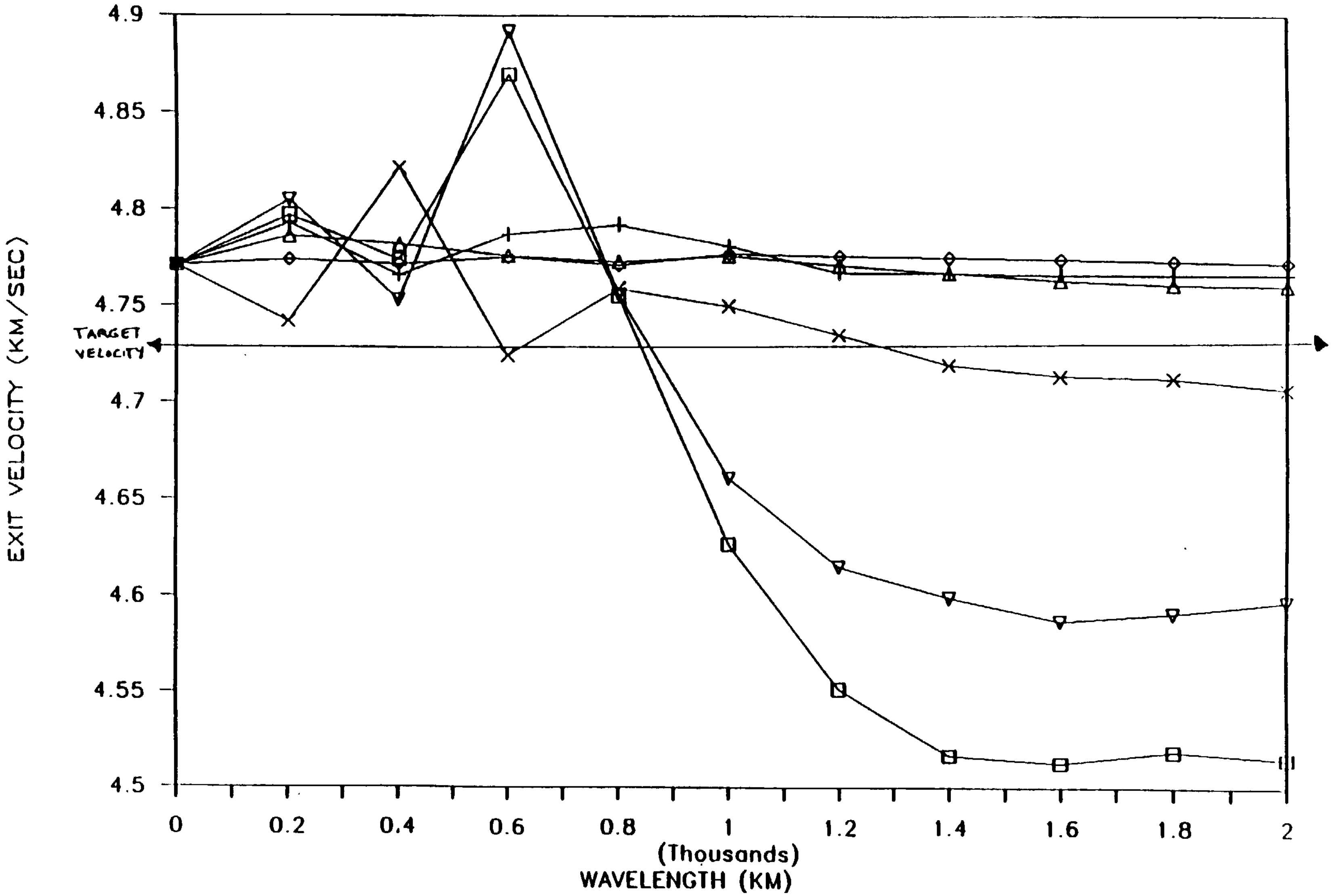
  DENS = DENS*[1.0 + 0.05*SQRT (DENS30/DENS)*SIN((Y/W)*2Pi)]

- Horizontal sine-wave density and vertical density-ratio and sine-wave density scaling

  DENS = DENS*[1.0 + 0.05*SQRT (DENS30/DENS)*SIN((Y/W + H/LZ)*2Pi)]

Using the COSPAR low atmosphere with sine wave distribution of the density, calculations were made which illustrate that for sine wave lenghts greater than 1000km the exit velocity errors are higher than for the lower wave lengths. The 30 - 40 km altitude region is by itself the most critical region.

EFFECTS OF HORIZONTAL DENSITY SINE-WAVE FORMS
ON AEROCAPTURE EXIT VELOCITY

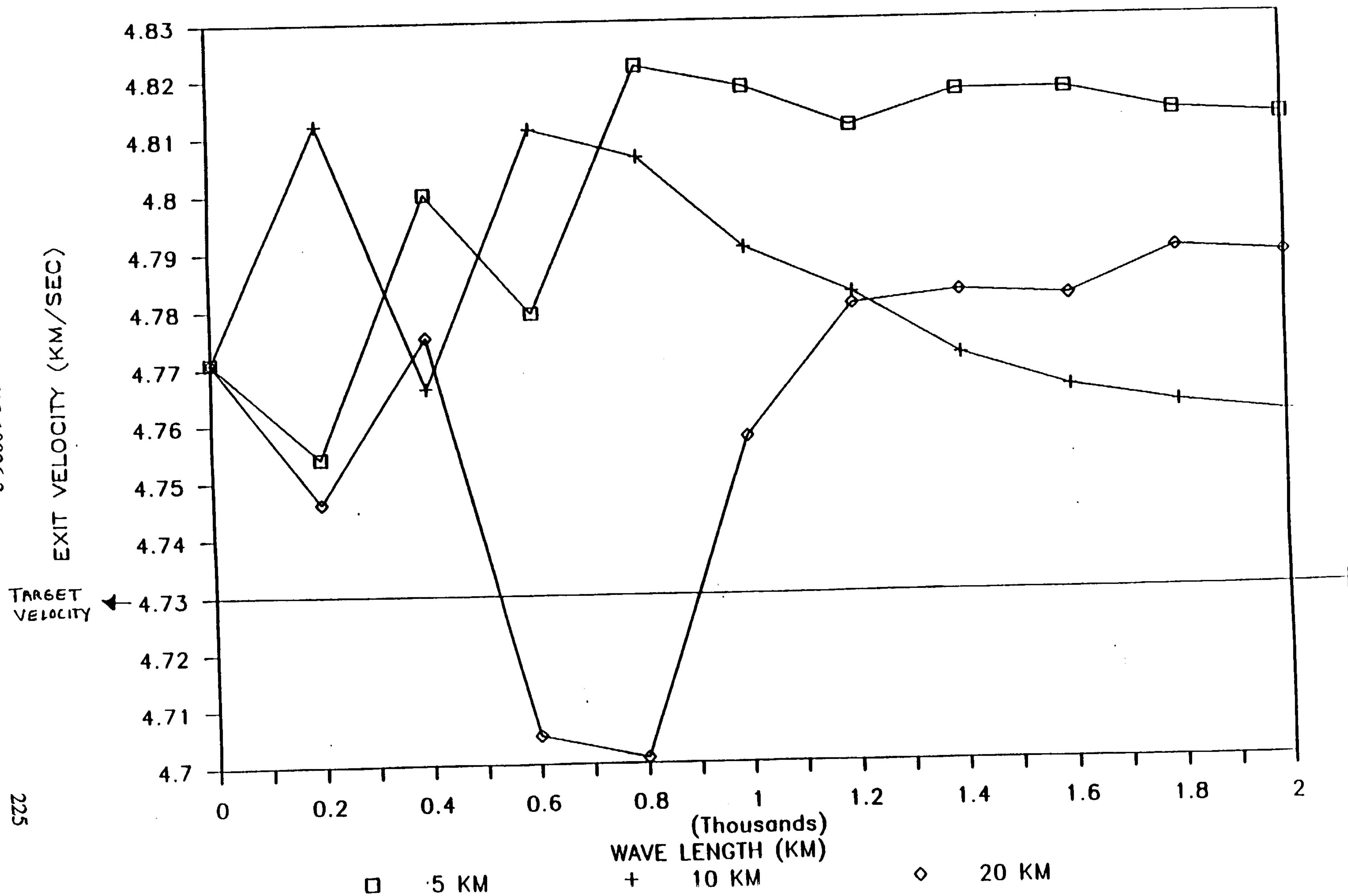


REGIONS IN WHICH WAVE STRUCTURE IS ACTIVE:

□	0 TO 100 KM ALTITUDE	△	50 TO 60 KM ALTITUDE
+	70 TO 80 KM ALTITUDE	X	40 TO 50 KM ALTITUDE
◇	60 TO 70 KM ALTITUDE	▽	30 TO 40 KM ALTITUDE

Guided trajectories were simulated for horizontal and vertical wave lengths. The horizontal wave length varied to 2000 km with vertical wave lengths of 5, 10, and 20 km. The larger vertical wave length of 20 km provides a more favorable atmospheric condition as the density variation in the vertical direction does not vary as much as the lower wave lengths in the critical region.

HORIZONTAL SINE-WAVE DENSITY AND VERTICAL DENSITY-RATIO AND SINE WAVE DENSITY SCALING



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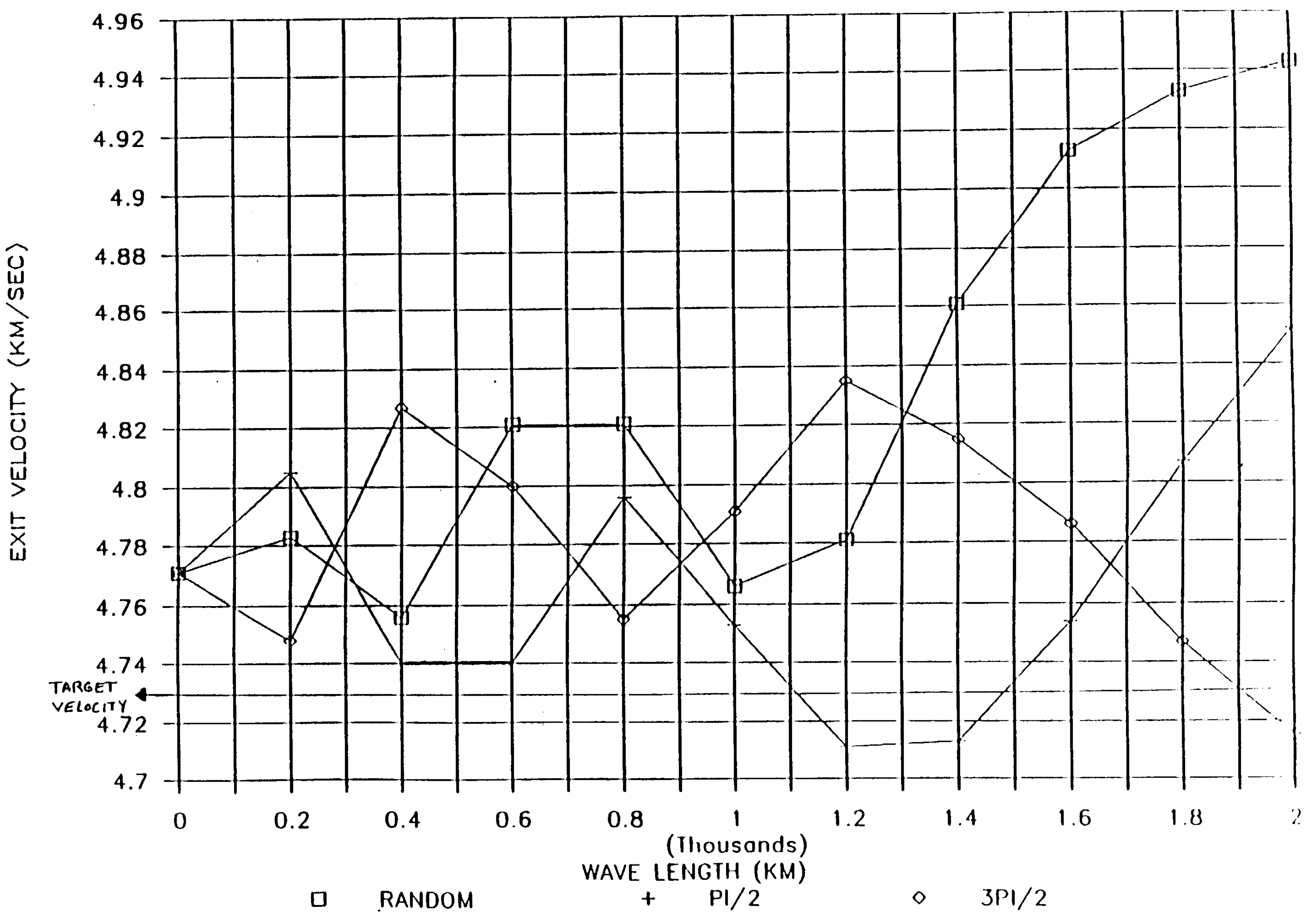
SCALING WITH SINE CURVE PHASE SHIFT



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Aeroheating Methods and Assumptions

Aeroheating estimation methods we are using are summarized here.

ADVANCED CIVIL SPACE SYSTEMS

Aeroheating Methods and Assumptions

BOEING

STCAEM/stl/15Mar90

Stagnation Point Heating

- **Radiative (Park Method)**
  - Equilibrium (Stagnation Pressure > 0.1 atm)
  - Optically thick gas (absorptivity = 0.5)
  - Park Method reliable to within ± 30%
- **Convective**
  - Modified Fay-Riddell
  - Fully Catalytic

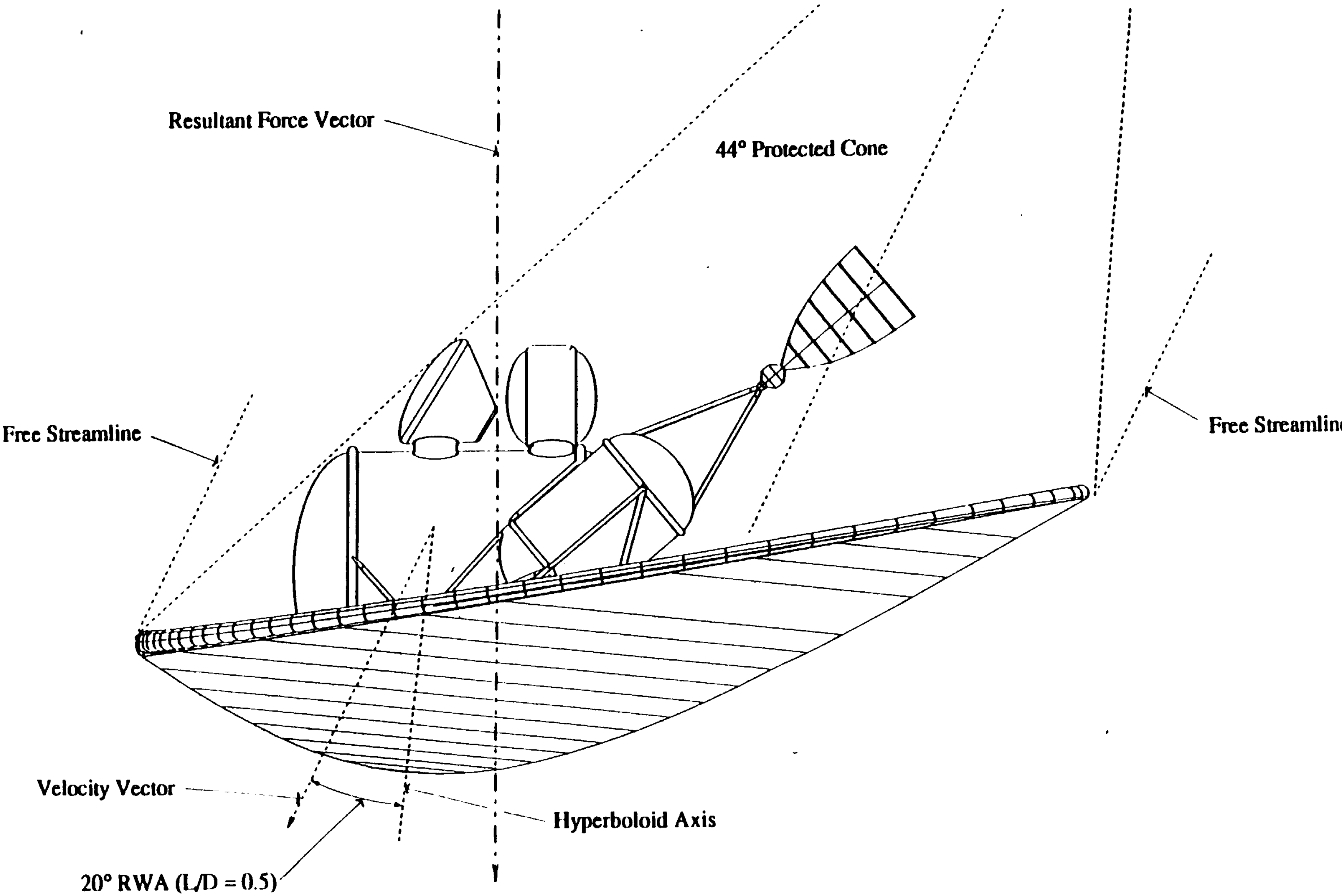
Distributed Heating(Continuum flow)

- **Radial Streamlines Assumed**
- **Radiation**
  - Approximate shock shape used.
  - Averaged Normal Velocity Component is the used in the Park Method
- **Convective (Boundary Layer Analysis Program)**
  - Axisymmetric Analog
  - Pressure Distribution: Newtonian Impact Theory
  - Laminar Flow (Re transistion = 2×10^6)

MTV Aerobraking Constraints

The aerobraking constraints applied to the MTV configuration are summarized on the facing page. These constraints include center of mass location for trim at the desired L/D, keeping the MTV itself in the protected wake region.

MTV Aerobraking Constraints



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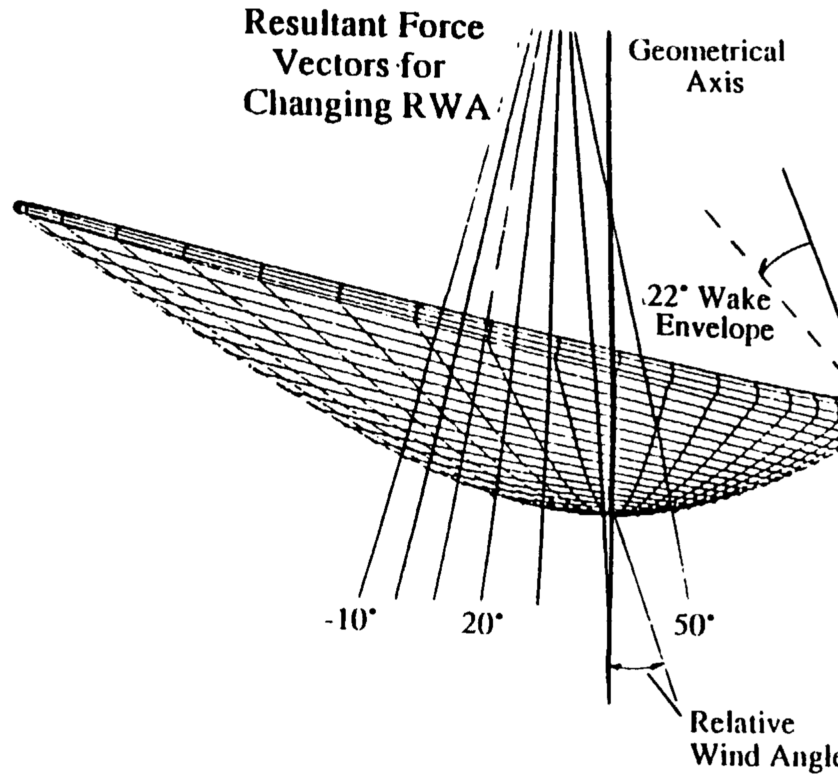
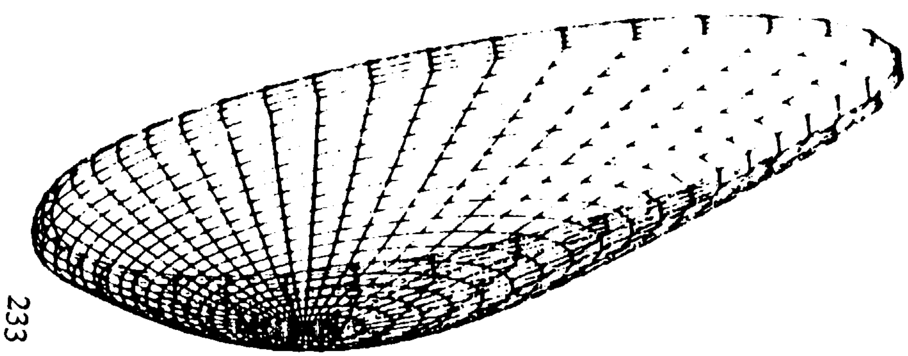
0.5 L/D AEROBRAKE

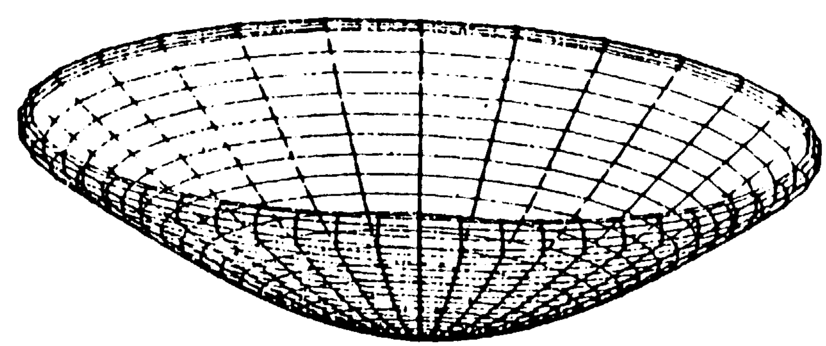
Values for Shape Parameters

Semimajor Axis Ratio	2.00
Eccen. of Body of Revol.	1.60
Eccen. of Cutting Cyl.	0.40
Truncation/SMA ratio	0.00
Lip Radius/SMA ratio	0.05
Lip Taper Ratio	0.30
Normalized x length	4.07
Normalized Plan Area	11.8
Normalized Surface Area	15.4

Angle	CL	Cd	L/D	Moment Arm
-10	0.1558	1.5251	0.1021	0.9105
0	0.3307	1.3956	0.2370	0.7861
10	0.4434	1.1902	0.3725	0.6489
20	0.4790	0.9434	0.5077	0.4913
30	0.4439	0.6983	0.6356	0.3042
40	0.3614	0.4867	0.7425	0.0795
50	0.2622	0.3215	0.8155	-0.1729







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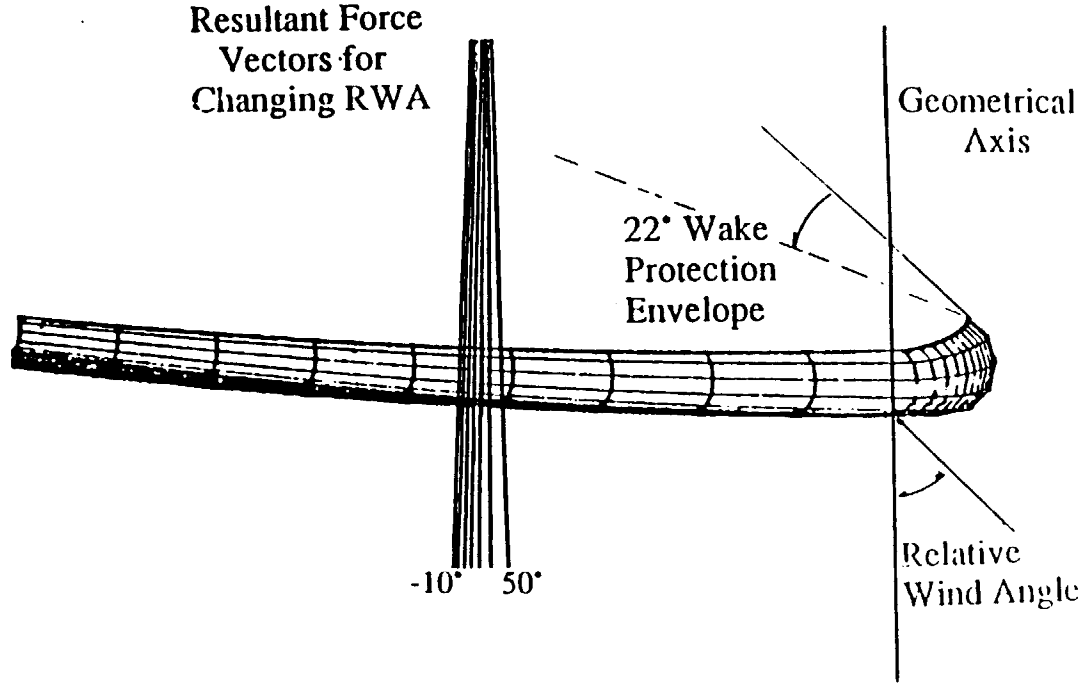
High L/D Aerobrake - Aerodynamic characteristics of the high L/D aeorbrake, based on Modified Newtonian impact theory

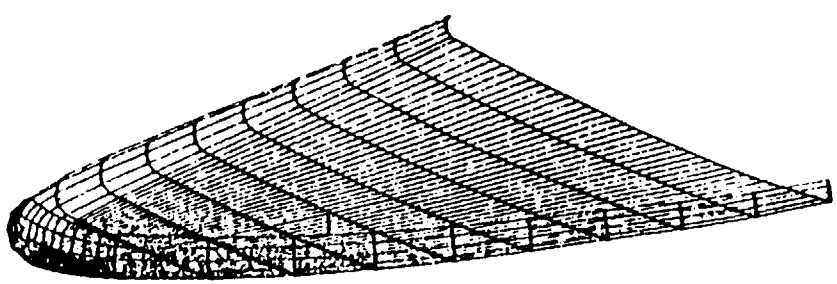
HIGH L/D AEROBRAKE

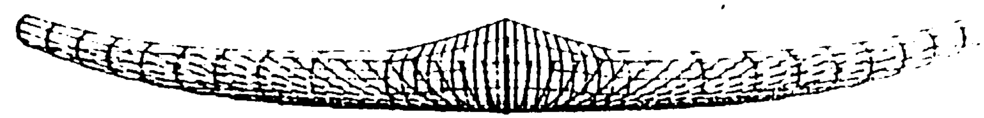
'alues for Shape Parameters

;emimajor Axis Ratio	2.00
;ccen. of Body of Revol.	3.00
;ccen. of Cutting Cyl.	1.03
'runcation/SMA ratio	0.50
ip Radius/SMA ratio	0.03
ip Taper Ratio	0.30
Normalized x length	1.120
Normalized Plan Area	0.851
Normalized Surface Area	0.988

Angle	CL	Cd	L/D	Moment Arm
-10	-0.2113	1.7644	-0.1197	0.3894
0	0.0926	1.7960	0.0516	0.3916
10	0.3759	1.6711	0.2249	0.3954
20	0.5800	1.4132	0.4105	0.4012
30	0.6682	1.0778	0.6199	0.4112
40	0.6311	0.7267	0.8684	0.4292
50	0.4974	0.4249	1.1706	0.4650

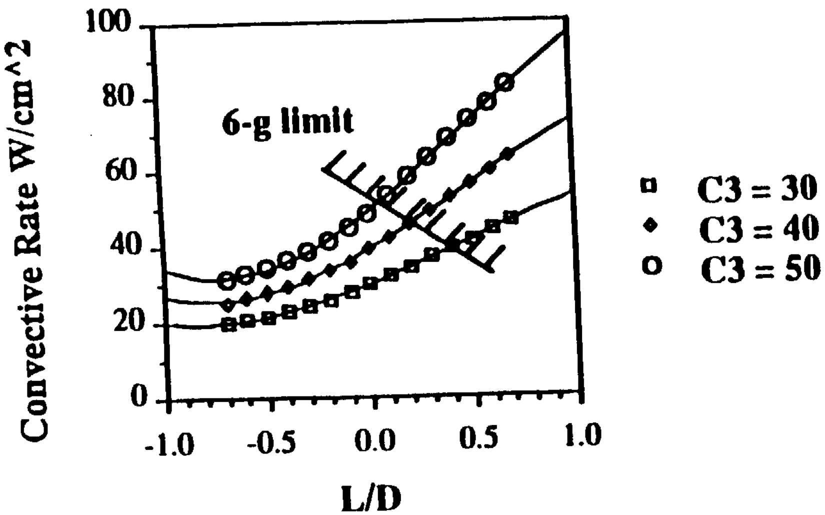


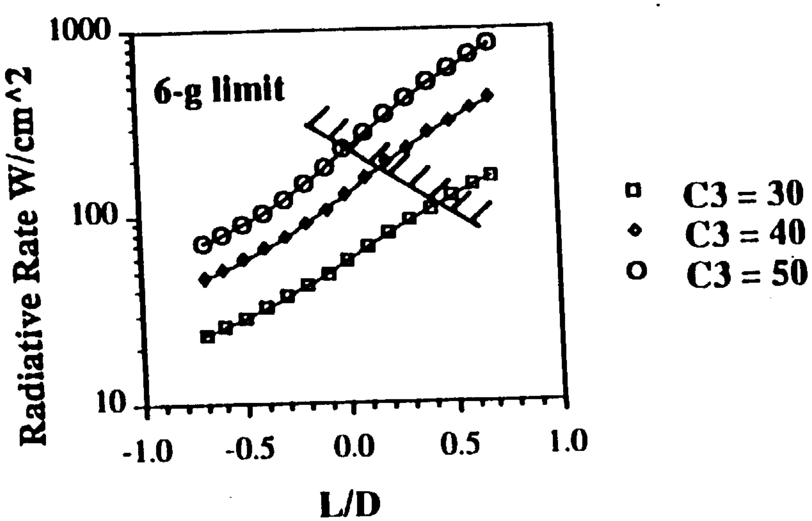


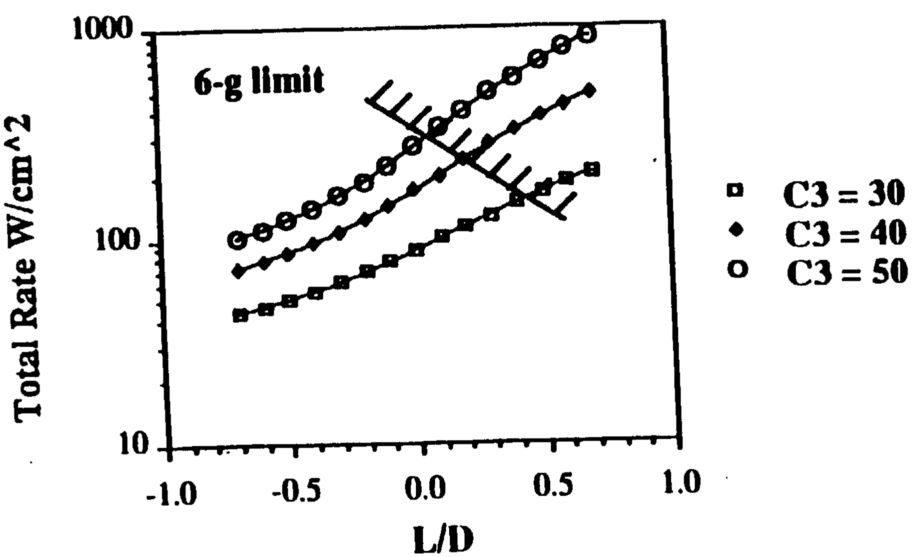


The stagnation point heating rates were calculated using the MARSIN code with a two dimensional trajectory and MarsGRAM atmosphere (high density). Te convective heat transfer was calculated for a fully catalytic wall, the radiative heat transfer was calculated using the Tauber–Sutton method and equilibrium flow. The stagnation point radius of curvature was 13m.The range of L/D varied form 1.0 to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative the heating rate decreases.

Mars Aerocapture - MTV







Stagnation Point
Radius of 13 m

Ballistic Coefficient
394 Kg/m²

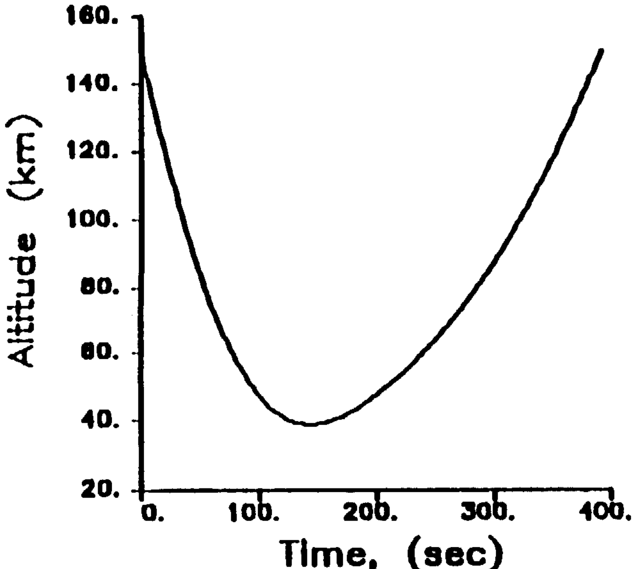
Radiation:
Tauber-Sutton Method

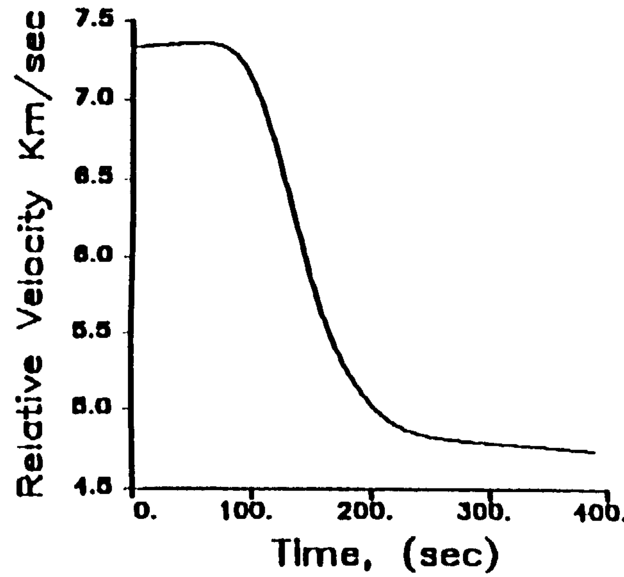
MarsGRAM
Atmosphere Hi
Density

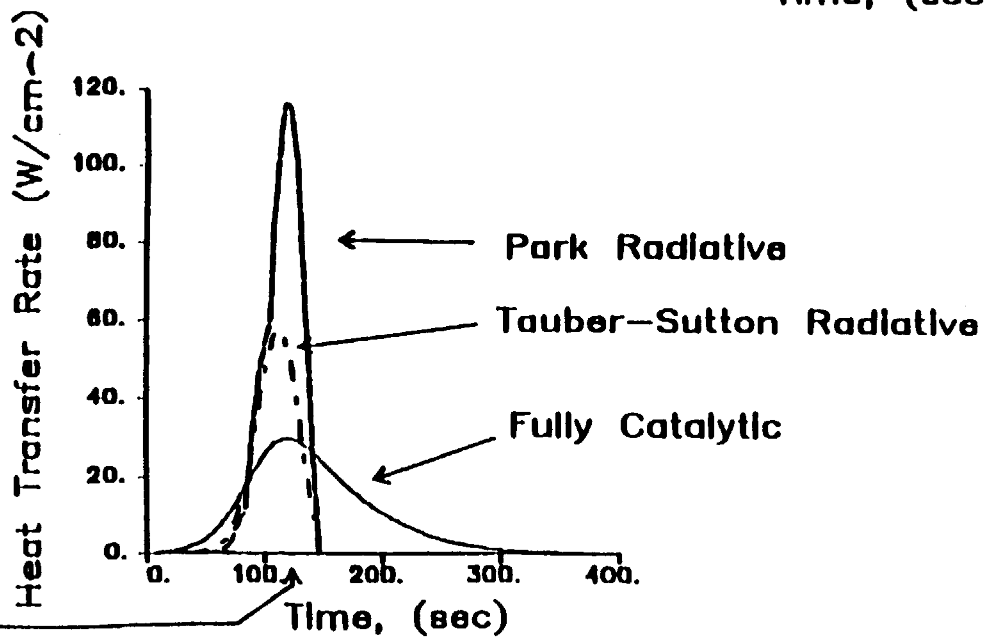
Mars Aerocapture Stagnation Point Heating - MTV

Aerocapture trajectory was computed with the MARSIN code.
Initial conditions are given as follows:
- MTV hyperboloid aerobrake
- L/D = 0.5
- Flight averaged L/D = 0
- Approach C3 = 30 sq km/sq sec
- Entry altitude = 150 km
- Entry velocity = 7.4 km/sec
- Ballistic coefficient = 394 kg/sq m
- MarsGRAM high density (winter solstice) for 2016

Heating - MTV







Maximum Total Heat Flux at 120 sec

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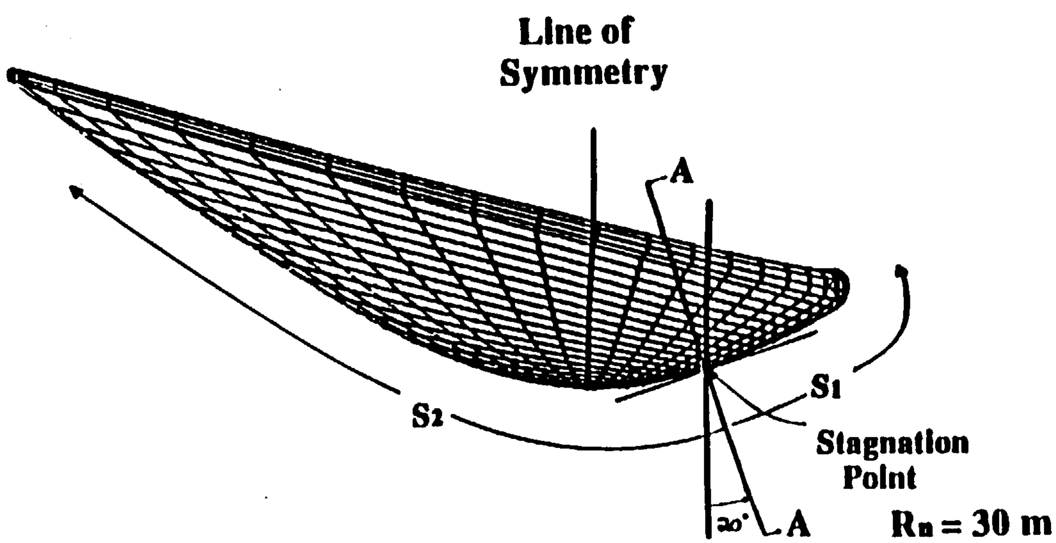
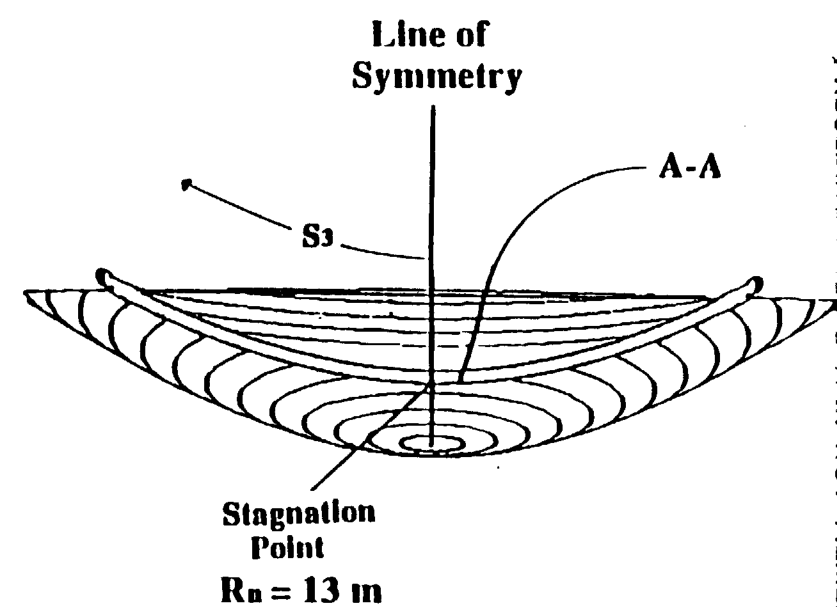
Streamlines for Hyperboloid Aerobrake

Assumed Streamlines S_1, S_2, S_3

For an angle of attack of 20°

L/D = 0.5





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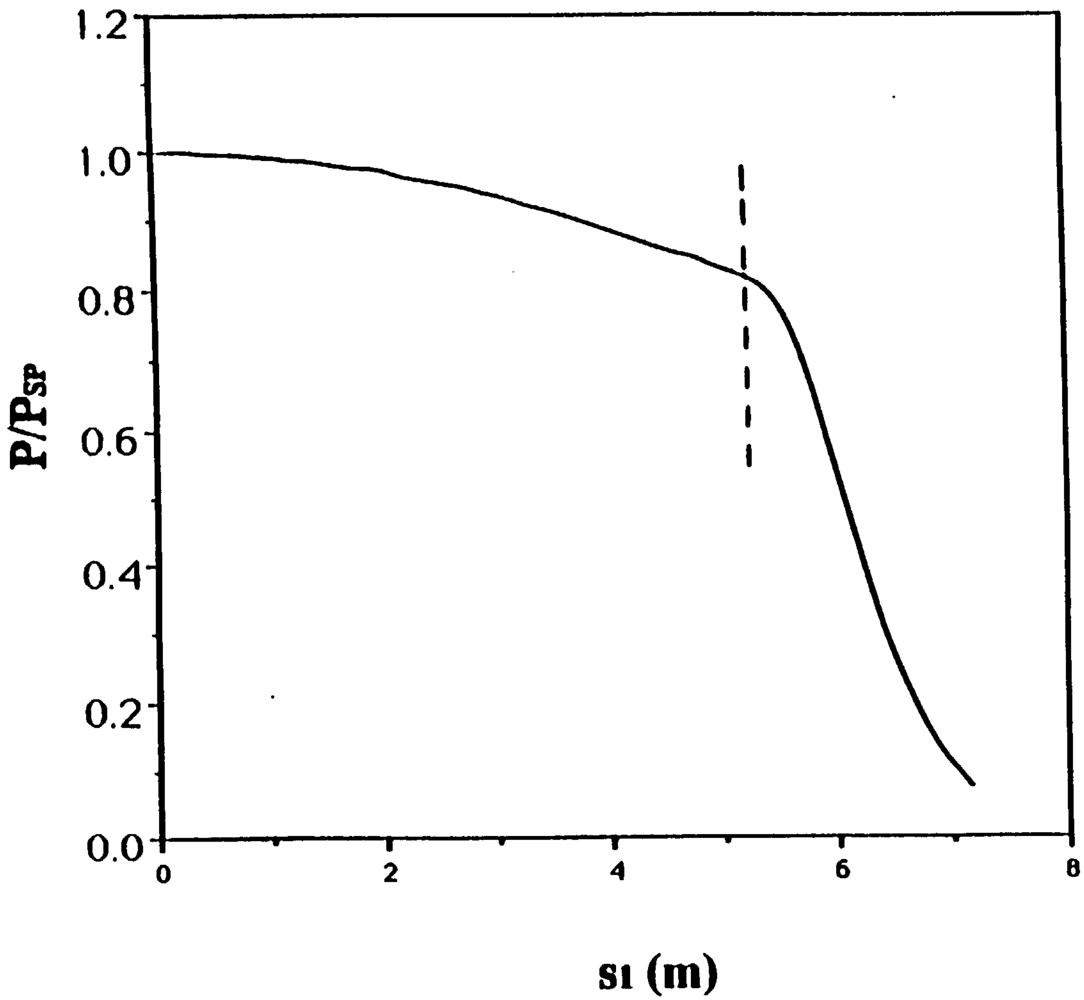
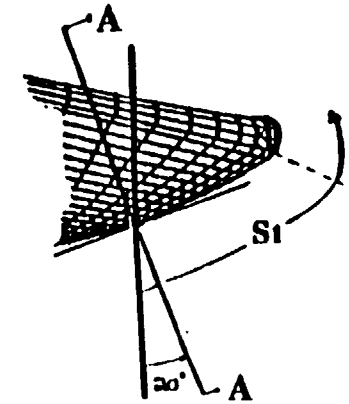
The following three charts show the pressure distribution along the fore , aft, and side streamlines based on a modified Newtonian theory, and unswept cylinder theory for the cylindrical lip.

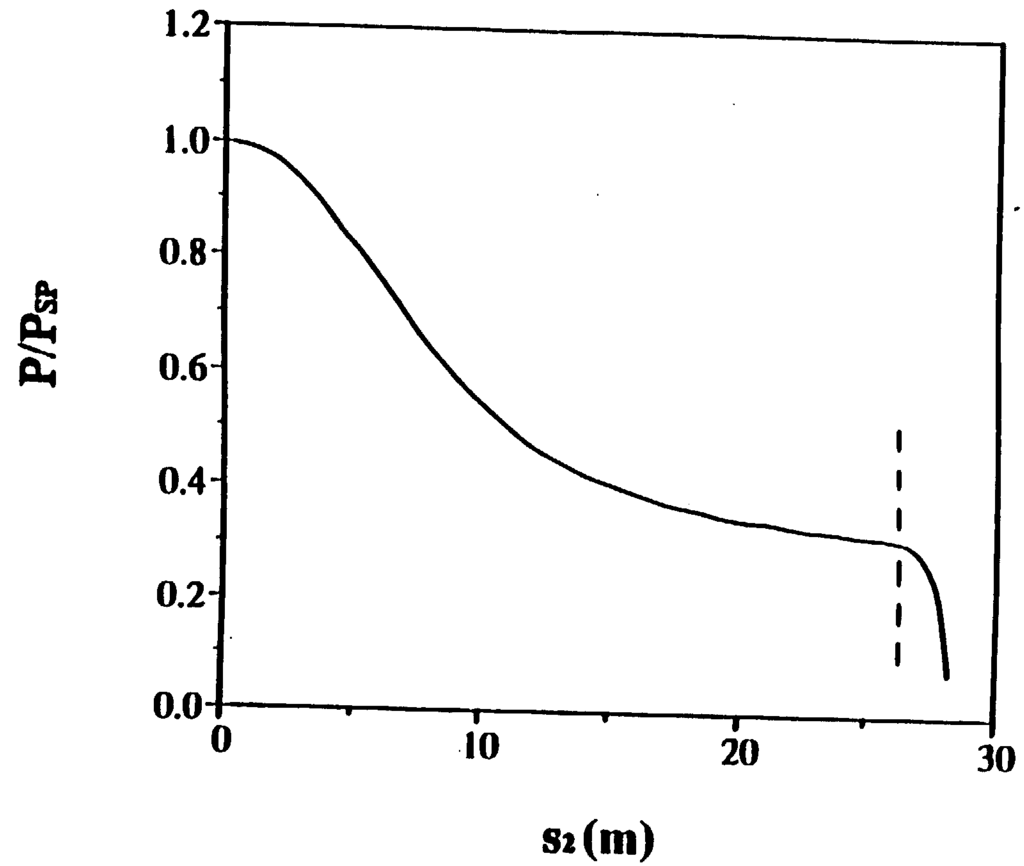
Distribution

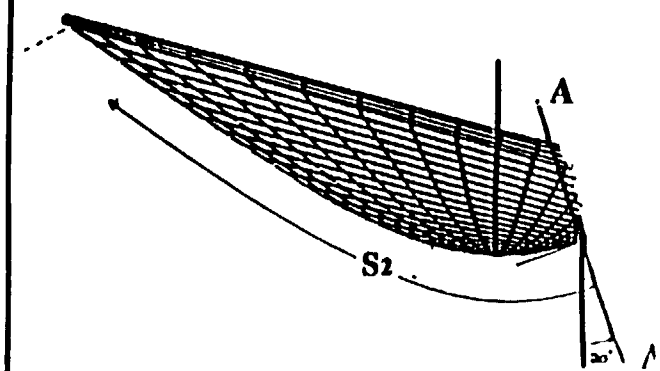




P_{SP} = Stagnation Point Pressure

Aft Centerline Pressure Distribution





P_{SP} = Stagnation Point Pressure

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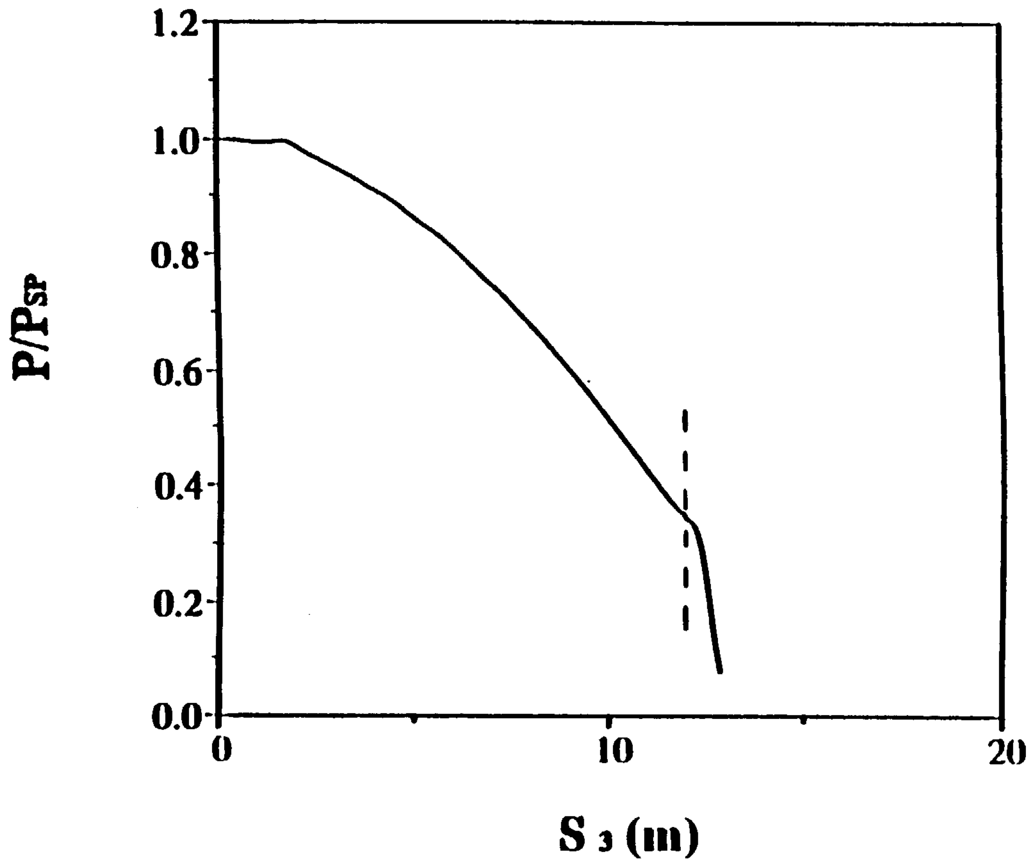
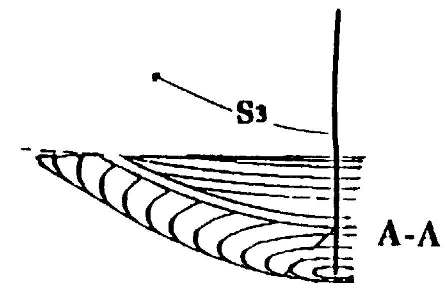
Side Streamline Pressure Distribution



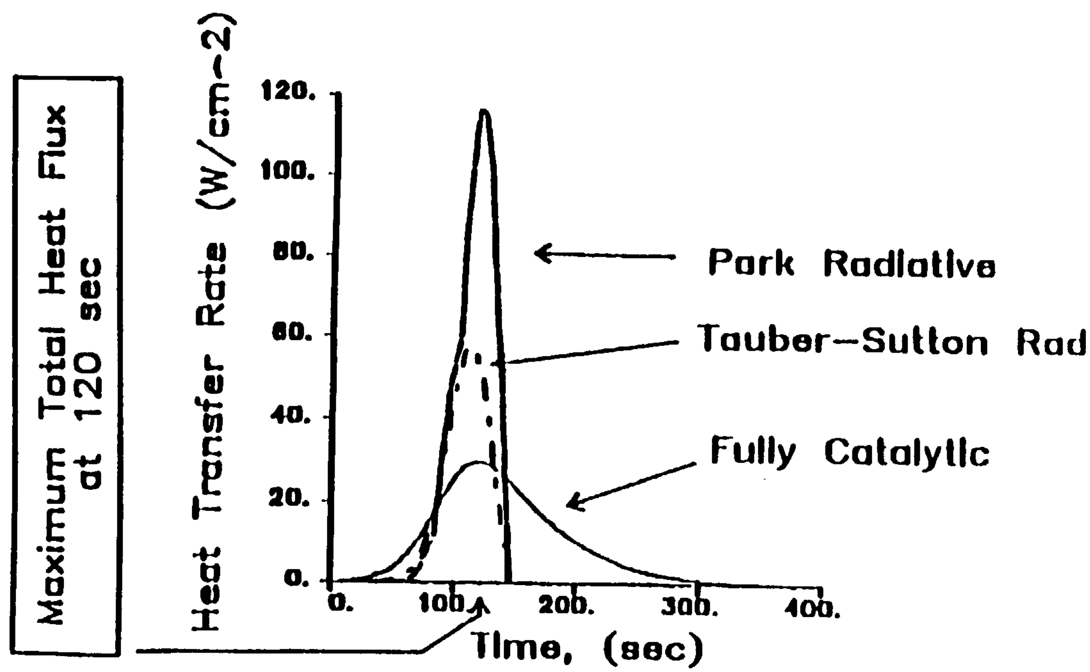


P_{SP} = Stagnation Point Pressure

The following chart delineates the aerothermal conditions at the stagnation point for maximum heating along the aerocapture trajectory.

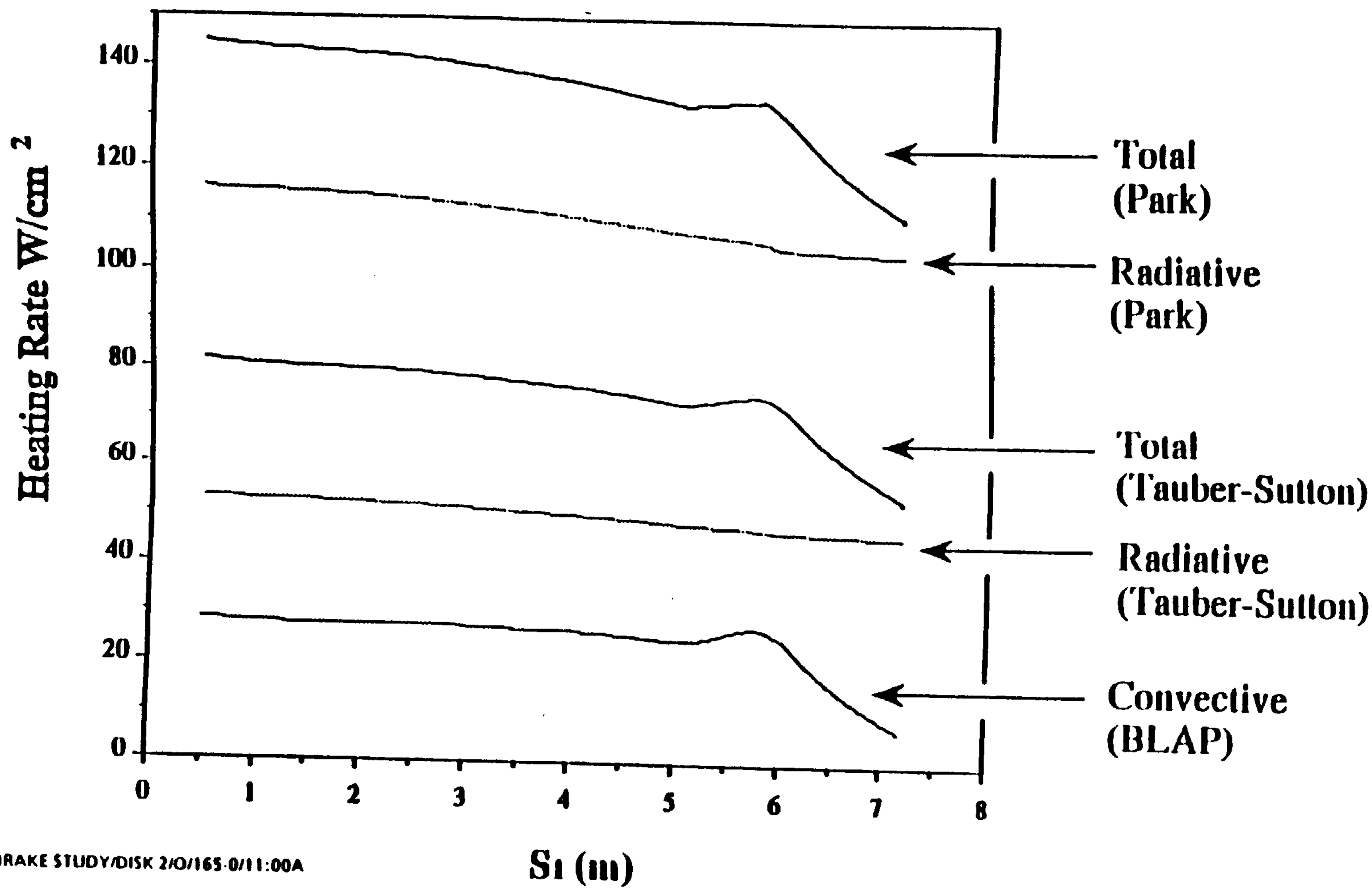
Conditions at Maximum Stagnation Point Heating

- Time - 120 sec
- Velocity - 6.714 km/sec
- Altitude - 41.2 km
- Density - $4.78 \times 10^{-7} g/cm^2$
- Radiative
  - Park - 116 w/cm^2
  - Tauber-Sutton - 53 w/cm^2
- Convective - 30 w/cm^2
- Equilibrium wall temperature (emissivity = .8)
  - Park - 2300°k
  - Tauber-Sutton - 2070°k
- Maximum G-level = 3.0



For the hyperboloid aerobrake of approximately 30m length, the heating rate was calculated using the boundary layer analysis program (BLAP) for the convective heat transfer with the radiative heat transfer being calculated using the Park method and the Tauber-Sutton method. For the forward stream line heating, the radiative heat transfer using the Park method is approximately twice that of the Tauber-Sutton method. No turbulent transition was assumed for the calculation. The heating rate at the stagnation point is approximately 146 w/cm^2 using the Park method and approximately 80 w/cm^2 for the Tauber-Sutton method.

Forward Streamline Heating



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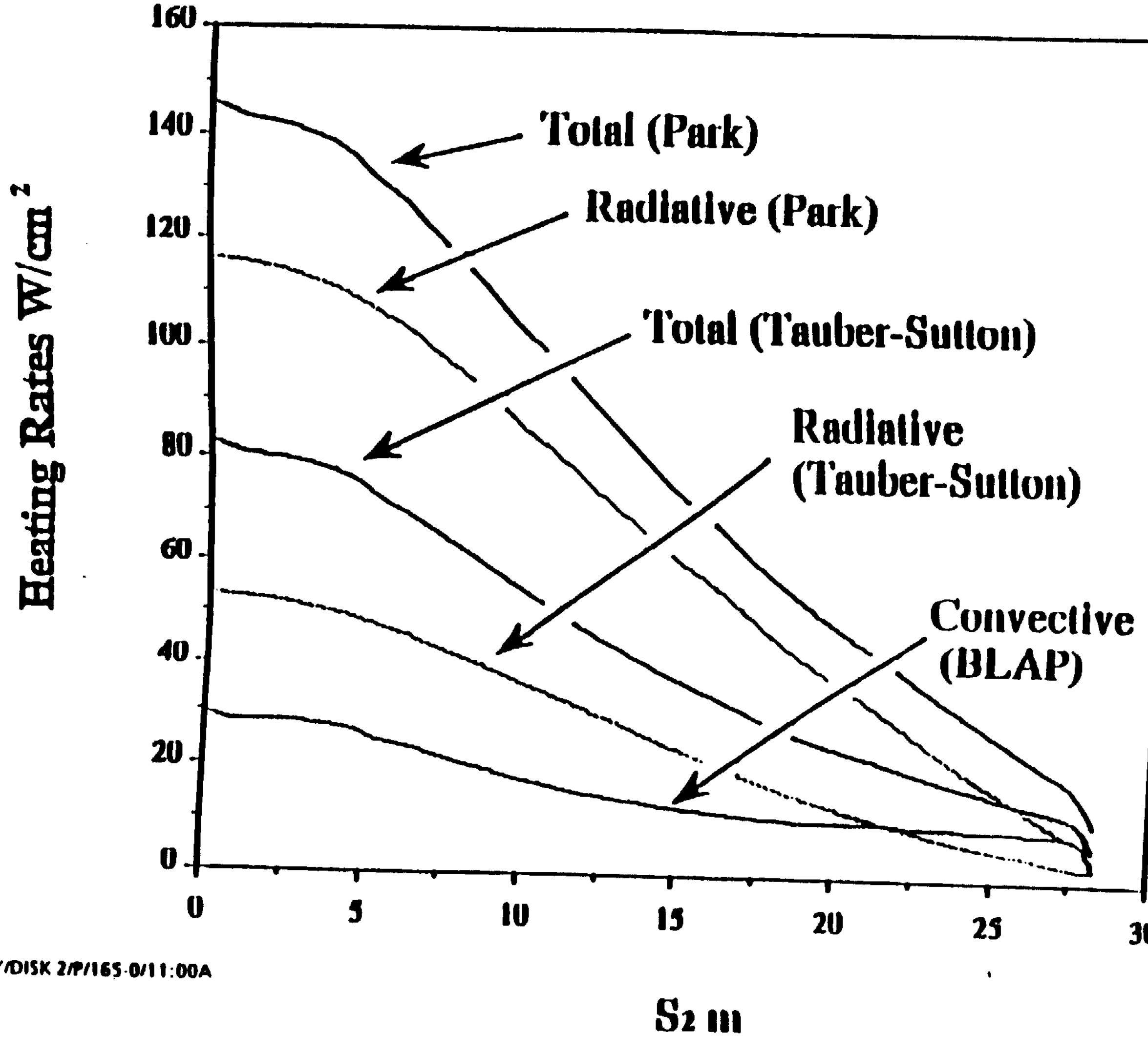
Aft Stre. nline Heating



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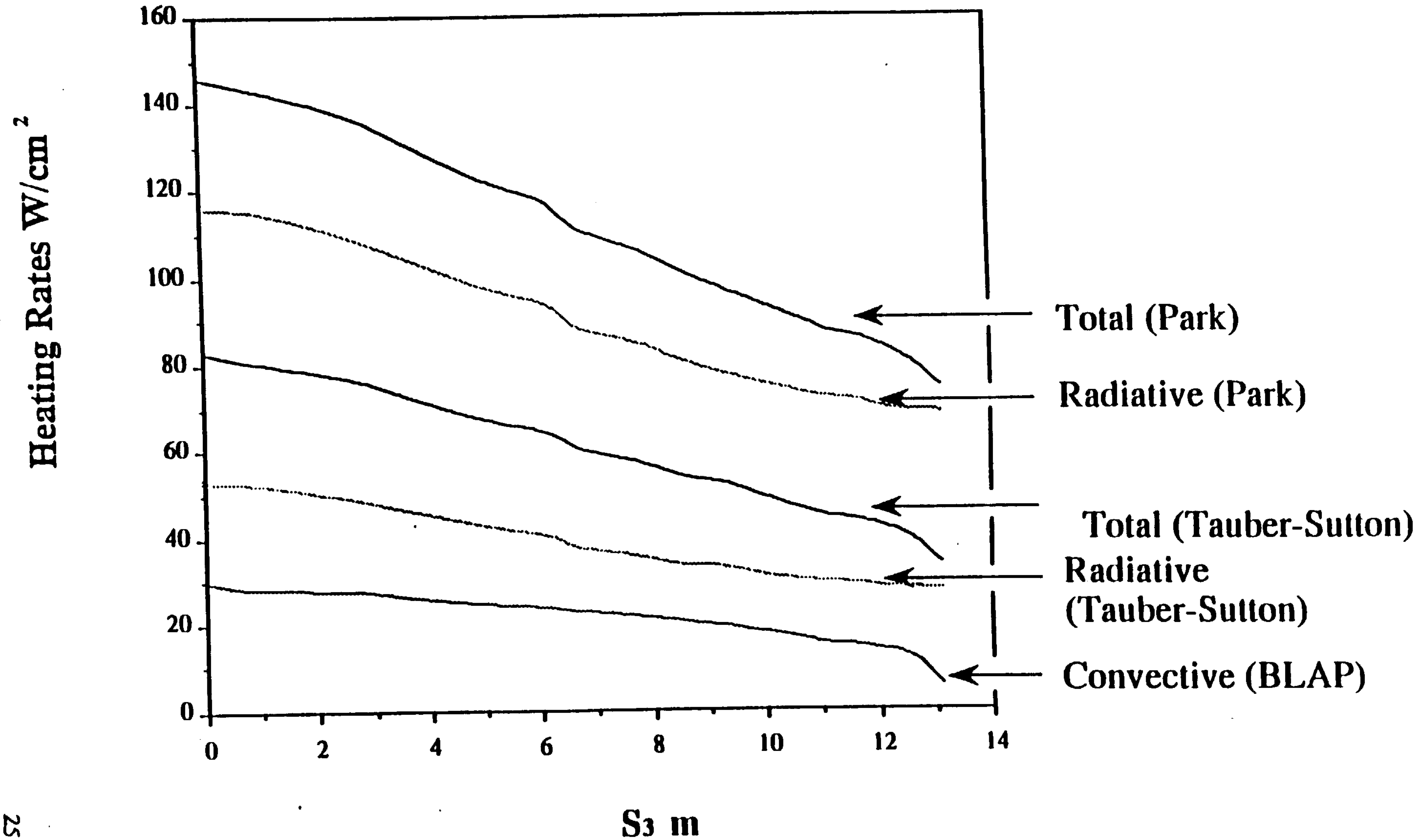
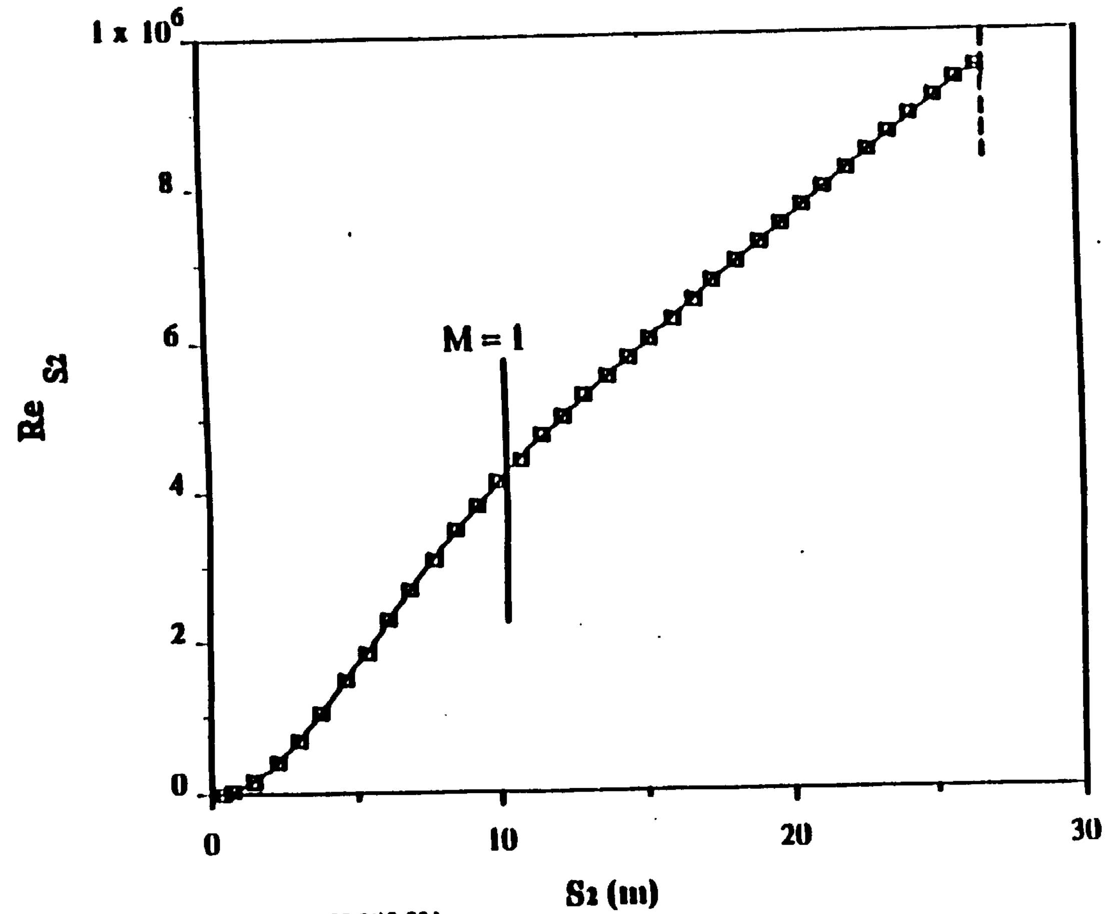
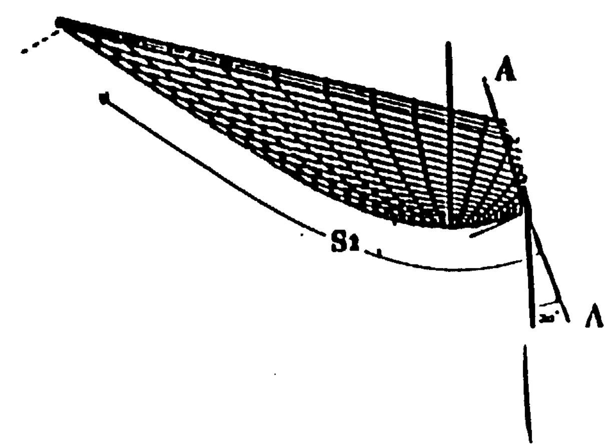
Side Streamline Heating



The Reynolds number was calculated for the aft stream line of approximately 30 meters. The Reynolds number was calculated based upon conditions behind the shock at the edge of the boundary layer utilizing the BLAP program. The calculations are for the highest heating rate at 120 sec. with a velocity of 6.7 km/sec at an altitude of 41.2 km. and a C3 of 30 km^2/sec^2. The local Reynolds number at the rear of body is less than one million.

Local Reynolds Number Along Aft Streamline





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Mars Transfer Vehicle
Incident Shock Layer Gas Radiation In the Base Region

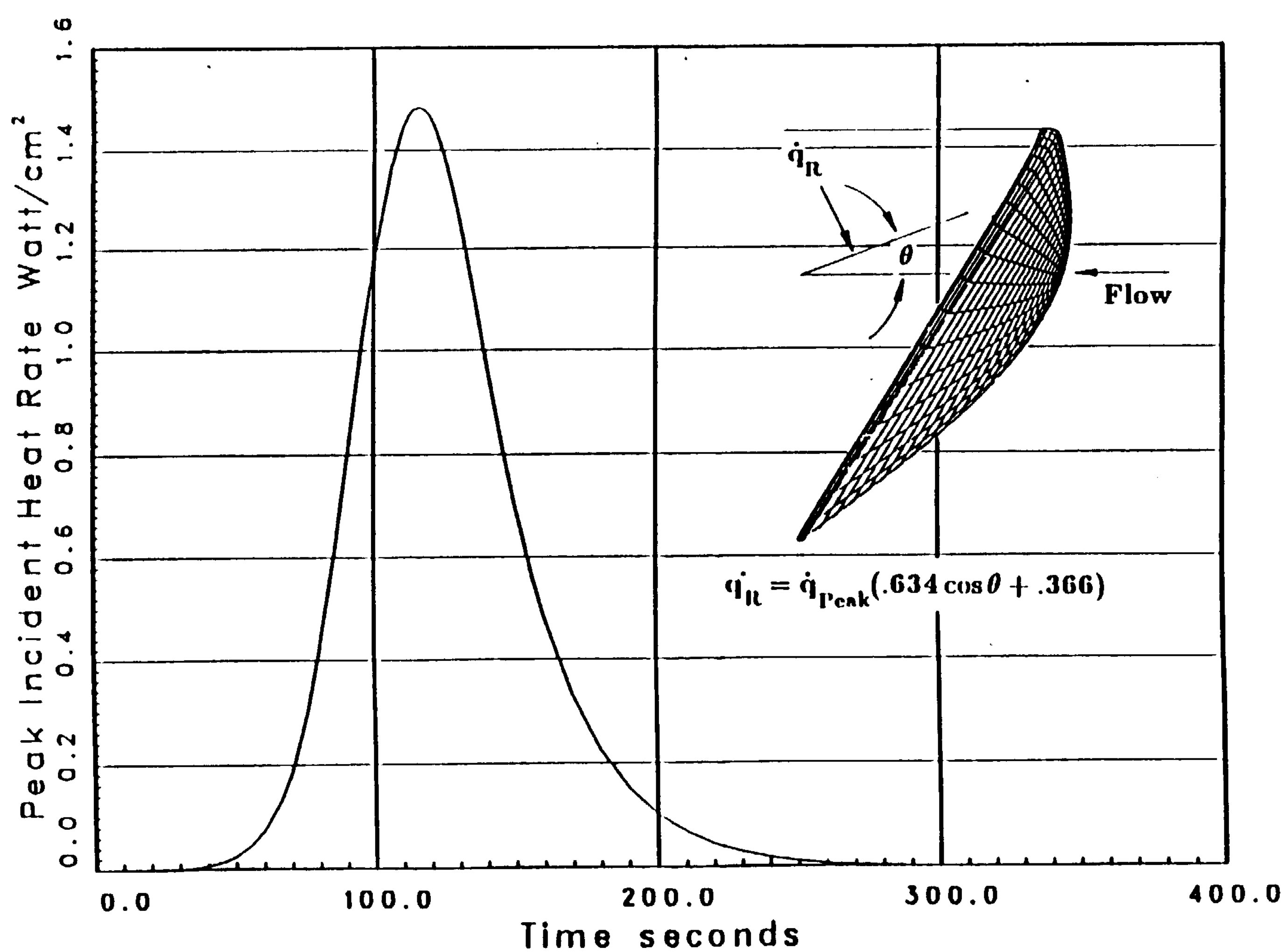
Preliminary results of work performed under subcontract, by RemTech Inc. to ACSS is displayed in the following three charts. The study involved examining the base flow heating regime, which includes both convective and radiative heating for a C3 of 30 km^2/sec^2 MTV aerocapture at Mars. The purpose was to determine the region of low heating behind the MTV and thus the protective cone for packaging the crew, habitat modules, and other cargo. The trajectory used for this analysis was provided by ACSS and is displayed on page 3-9, where the maximum stagnation point heating is 83 W/cm^2.

The graph below displays the base radiative heating rate as a function of time for the C3=30 trajectory. The equation shown on this chart gives the radiative heating rate to a surface in the base, for varying view angles. These radiative heating predictions are based on relationships derived and used for the AFE base flow heating regime. The maximum base flow heating occured at 114 seconds with a maximum radiative rate of ~1.5 W/cm^2. This value is only 2.6% of the peak stagnation point value.

MARS TRANSFER VEHICLE

INCIDENT SHOCK LAYER GAS RADIATION IN THE BASE REGION

REMTECH



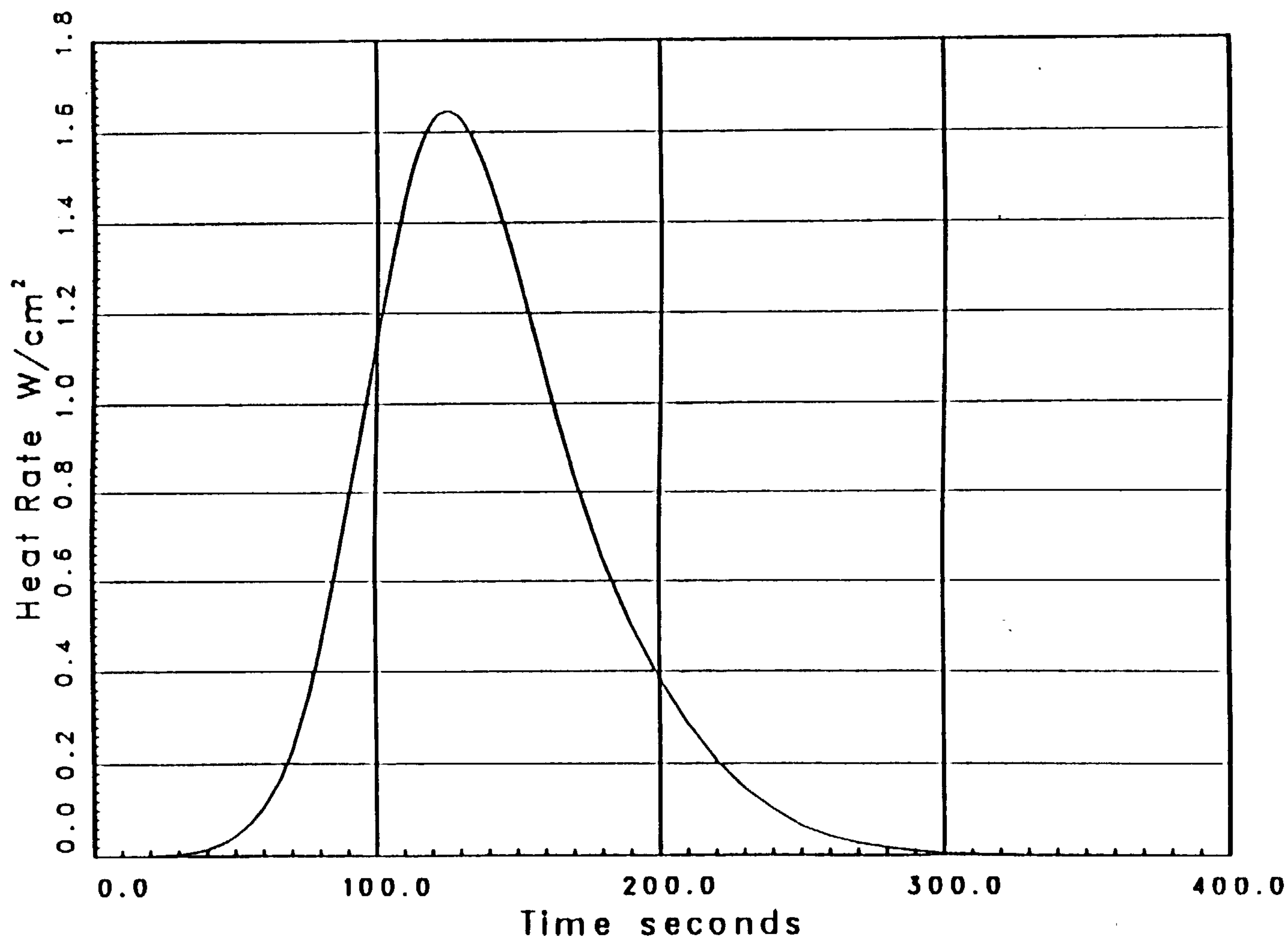
Mars Transfer Vehicle
Base Convective Heat Rates Tw = 1367

The graph below shows the convective heating rate within the base region along the C3=30 aerocapture trajectory. Maximum base convective heating occured at 114 seconds, with a value of ~1.7 W/cm^2. For the heating rate calculation, the assumed wall temperature (Tw) was 1367 K. Changing this Tw value by ± 30% resulted in only a ~1% change in heating rate and thus Tw was found to be of small importance in computing the heating rate. Total heating rates for the base region are on the order of 3.2 W/cm^2 , thus requiring a need for some TPS on surfaces within the base.

MARS TRANSFER VEHICLE

BASE CONVECTIVE HEAT RATES Tw = 1367 K

REMTECH

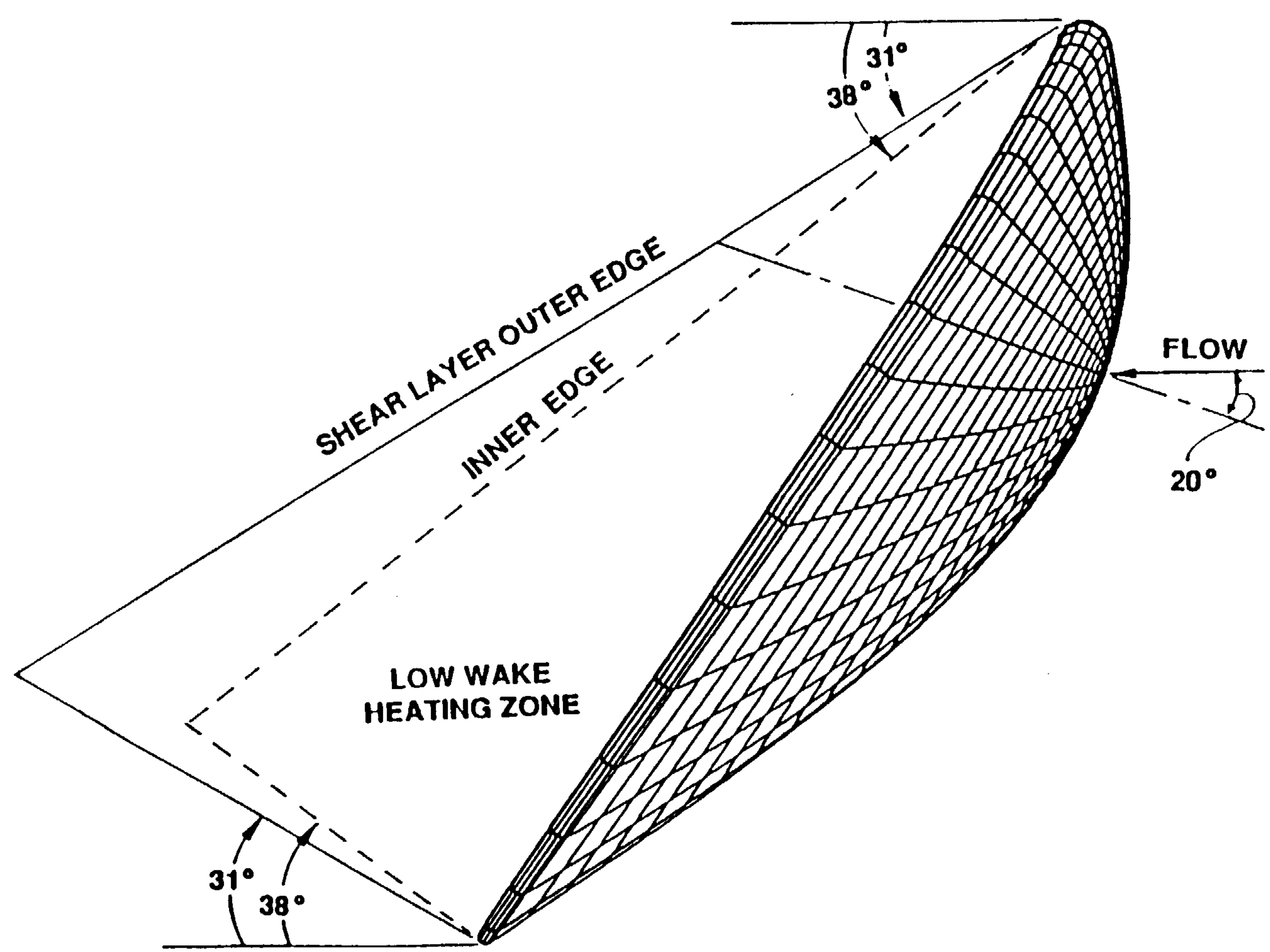


Wake Closure Zone at Peak Heating

Shown below is a graphical depiction of the wake closure, and protective low heating region for the MTV aerobrake. This shear layer angle is based on calculations made for the expansion of the flow around the lip of the MTV aerobrake. Flow field properties around the lip were estimated using the BLIMPK program. The outer edge of the shear layer was calculated to be at 31° from the flow direction. The viscous region, estimated from experimental data, accounted for an additional 7° resulting in a total wake deflection angle of 38°. This preliminary estimate of the wake flow would impact the packaging of the aerobrake contents.

WAKE CLOSURE ZONE AT PEAK HEATING

REMTECH



The following approximate temperature contours are based on the BLAP convective and Tauber_Sutton radiative streamline heating distribution.

Tauber-Sutton Radiation

Trajectory

- C3 = 30 km^2/sec^2
- Flight Averaged Lift = 0
- L/D = 0.5
- Ballistic Coefficient = 39.4 g/cm^2
- Max G-level = 3.0

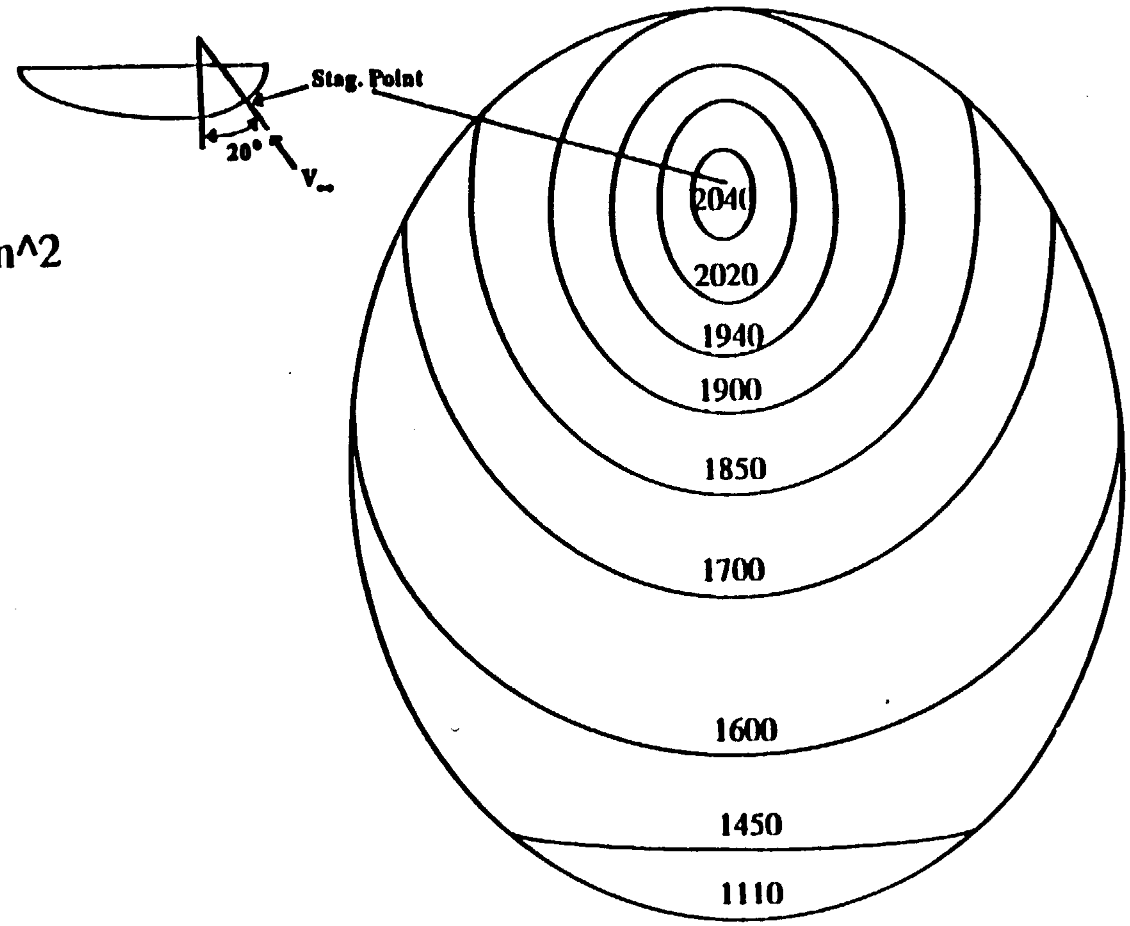
Conditions at Max Heat Rate

- Time = 120 sec
- Velocity = 6.714 km/sec
- Altitude = 41.2 km
- Density = 4.78x10^-7 g/cm^2
- Total Heat Rate = 83 w/cm^2
- Stagnation Temp. = 2069 K

Total Heat Load

- Q = 5,880 j/cm^2

Emissivity = 0.8



Temperature in K

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ADVANCED CIVIL SPACE SYSTEMS

Preliminary Results

BOEING

Stagnation Point Heating (Park Method)

- Radiative heating for $30 < C_3 < 50\ km^2/sec^2$
  $\geq$ 80% of total (Park)
  $\geq$ 65% of total (Tauber-Sutton)
- Stagnation temperatures for $C_3 \geq 30\ km^2/sec^2$ exceed 2000° k

	Park		Tauber-Sutton	
C3	Q W/CM2	T °K	Q W/CM2	T °K
30	146.	2383	83.	2068
40	299.	2850	170.	2474
50	481.	3210	274.	2790
1993 technology ~	68.	1968		

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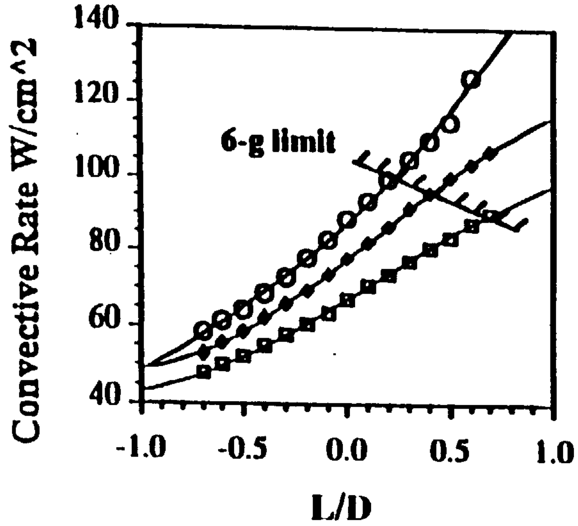
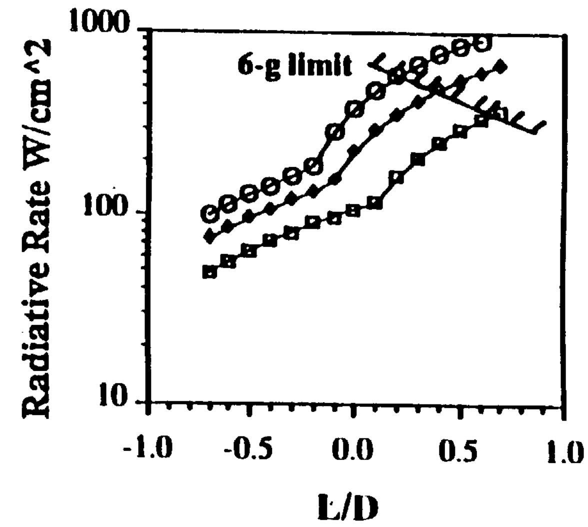
GRM/2H895/AEROBRAKE STUDY/DISK 1/J/145-0/2:00P

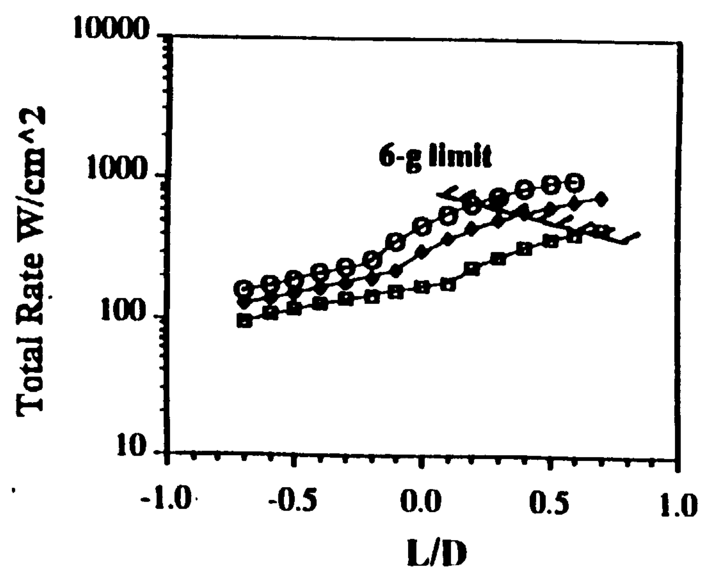
The stagnation point heating rates were calculated using the MARSRT*code with a two dimensional trajectory. The convective heat transfer was calculated for a fully catalytic wall; the radiative heat transfer was calculated using the Park method and equilibrium flow. The stagnation point radius of curvature was 13 m. The range of L/D varied from 1.0. to -1.0. For each condition a fixed L/D was used. The calculations illustrate that as L/D becomes negative, the heating rate decreases.

***MARSRT - (mars return) an aerocapture at Earth heating code**

Earth Aerocapture - MTV





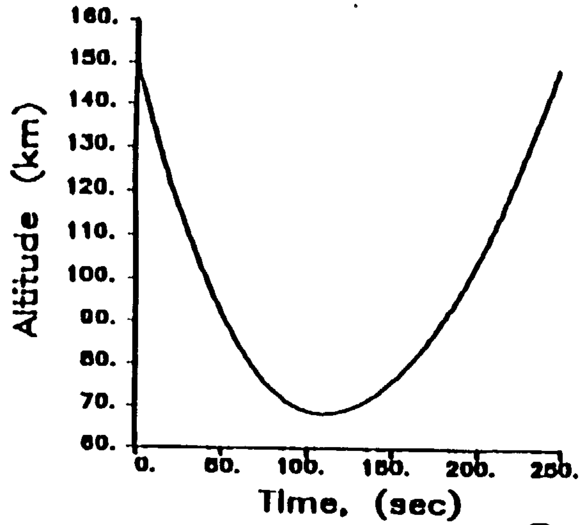


Stagnation Point
Nose Radius 13 m

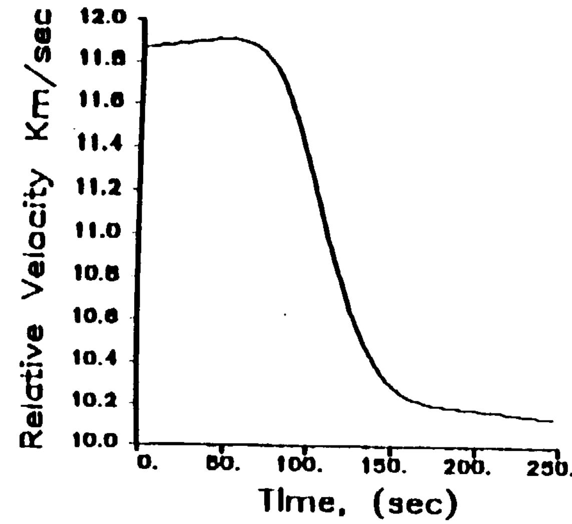
Ballistic Coefficient 225 kg/m^2

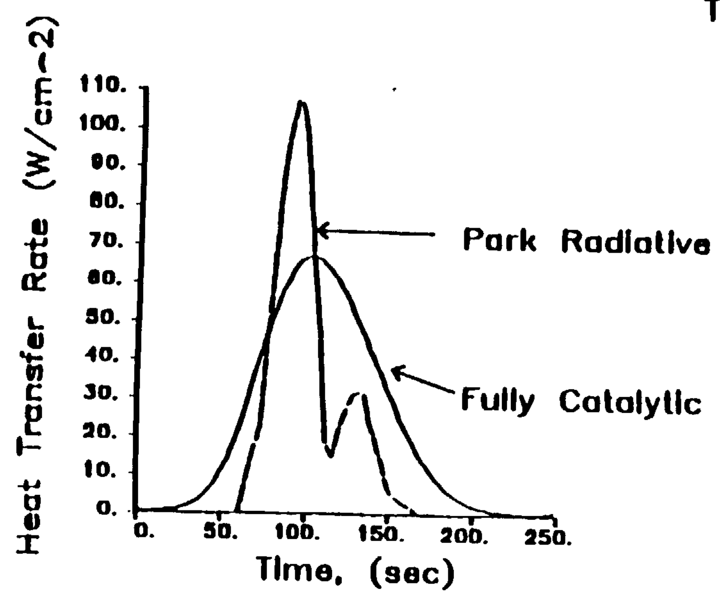
Radiative Heating
Park Method

For an average L/D of 0.5 using the same relationships as previously, the Earth aerocapture total heating rate is 172 w/cm^2 using the Park method. The peak convective heating rate was about 70 w/cm^2 calculated by the BLAP code. The earth aerocapture stagnation heating rate is higher than the stagnation point heating rate for Mars which was calculated to be 146 w/cm^2.



Hyperboloid 0.5
L/D Aerobrake





G-GRM/2H89S/AEROBRAKE STUDY/DISK S/N/165-0/10:00A

Two structural designs, spar and truss, were used for the hyperboloid aerobrake. The aerobrake has a length of approximately 31 meters, width of 28.2 meters and height of 6.5 meters. In both cases the aerobrake accommodates an engine hatch. In the case of the truss configuration, the tetrahedral truss respective points were projected on to the aerobrake surface to accommodate the required curvature. The spar configuration used a carbon magnesium metal matrix and the truss configuration used graphite epoxy. In both cases an aluminum honeycomb core was employed with a titanium face sheet. The Mars excursion vehicle had a payload of 81 metric tons and the Mars transfer vehicle the payload was 153 metric tons.

Design Assumptions:

- Constant spar cross-sections, curved profiles
- C/Mg metal matrix spars (density 1830 kg/cu. m.)
- Payload: Mars Excursion Vehicle, 81MT
- 6g maximum acceleration
- 8 payload attach points (4 frame and 4 landing leg points)
- Relative wind angle = 20 degrees
- Variable pressure distribution, range 1.5psi to 3psi
- Structure temperature = 394K (250F)
- Secondary spar pattern to be triangular for greater shear resistance

G-GRM/2H895/AEROBRAKE STUDY/DISK 4/I/165-0/11:00A

Structural Con ept - Spar Configuration

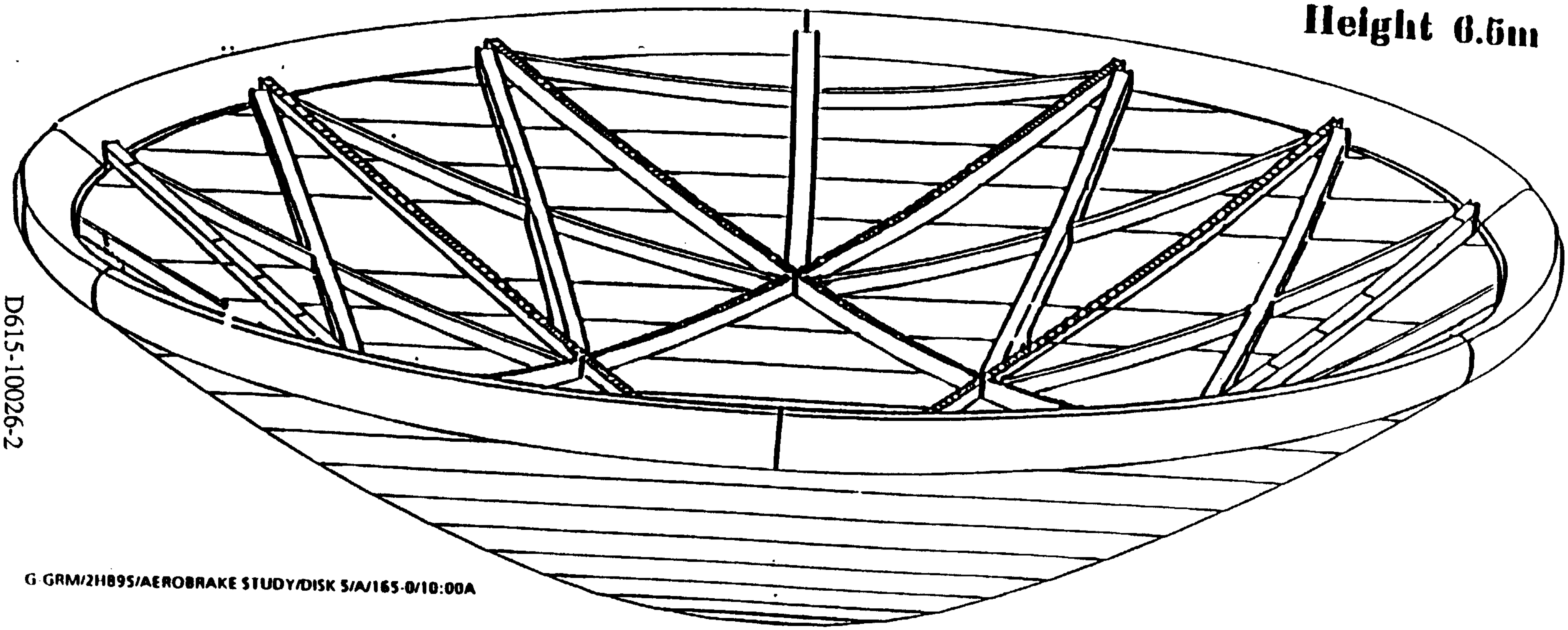
3D View

Size

Length 31.1m

Width 28.2m

Height 6.5m



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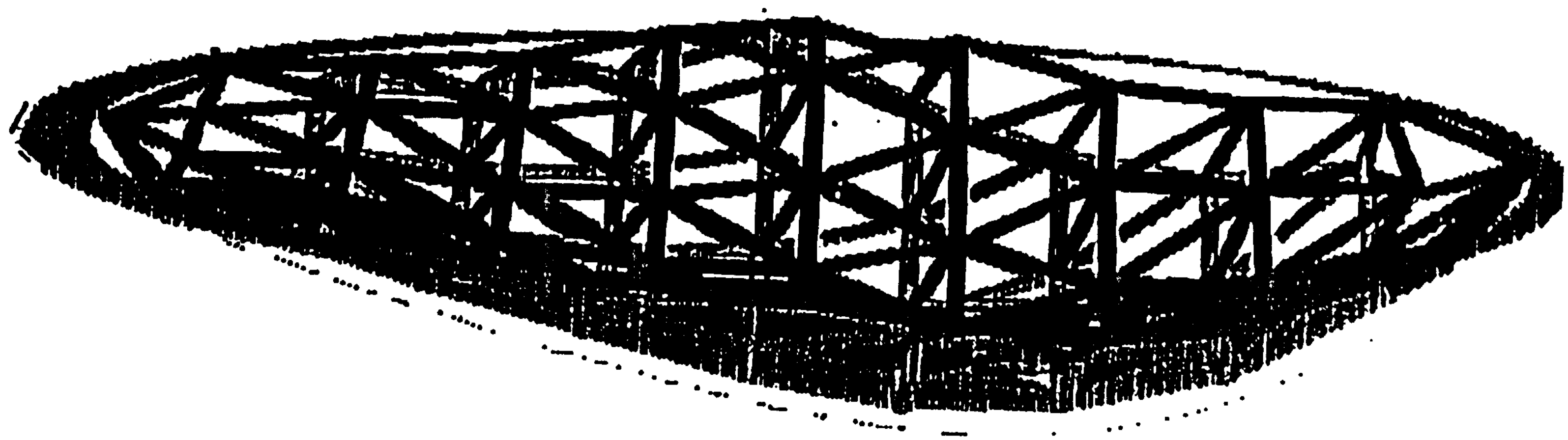
3D View



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Aerobrake Structural Design

81 mt payload, MEV

19.5 inch spar depths:	105 ksi spar strength		200 ksi spar strength	
Primary spar weight:	5,390 kg	(11,859 lb)	2,571 kg	(6,052 lb)
Secondary spar wt:	3,827 kg	(8,420 lb)	2,975 kg	(6,546 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	19,275 kg	(42,407 lb)	15,784 kg	(34,726 lb)
22.5 inch spar depth:				
Primary spar weight:	4,989 kg	(10,978 lb)	2,484 kg	(5,465 lb)
Secondary spar wt:	3,809 kg	(8,379 lb)	2,596 kg	(5,711 lb)
Honeycomb weight:	6,758 kg	(14,868 lb)	6,758 kg	(14,868 lb)
TPS weight:	3,300 kg	(7,260 lb)	3,300 kg	(7,260 lb)
Total aerobrake weight:	18,856 kg	(41,483 lb)	15,138 kg	(34,726 lb)

Note: 200 ksi option may require additional material technology development efforts.

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- Many state-of-the-art advances are needed to support on-orbit assembly and checkout
    - Robotics
    - Non-destructive inspection
    - "Smart" structure
    - Joint closure

- Advanced Thermal Protection System (TPS) materials are essential for $C3 > 30$; temperatures > 1800°C will probably require a mass penalty

G-GRM/2H895/AEROBRAKE STUDY/DISK 4/O/165-0/11:00A

Aeroheating Principal Findings

The findings from our current analysis are that Mars capture aeroheating is a significant problem, unless correlations that predict less radiative heating can be verified. Several work-arounds are noted on the chart. In the next three months, we will be working with alternative radiation correlations and exploring the efficacy of these work-arounds.

Aeroheating Principal Findings

- **Mars Aerocapture Radiation Problems**

  - For C3's from 30-50 km^2/sec^2, Radiative flux is 80-90% of total heat flux.

- **Stagnation Temperatures for C3's ≥ 30 are above near term reradiative technologies.**

eg.

C3	Q(w/cm^2)	T °K
30	146.	2383
40	299.	2850
50	481.	3210

Note: 1993 reradiative technology ≈ 68 w/cm^2 or ≈ 1968 ° K

- **Options**

  - Use of ablators for Mars Aerocapture.
  - Improve reradiative materials.
  - Limit the Missions to lower approach C3's.
  - Modify or Change the Aerobrake shapes.
  - Optimize Trajectories for miminal aeroheating (Down-Lift).

- **Needs**

  - Improved Analysis for Non-Equillibrium Radiation in CO_2 Atmosphere
  - Engineering Methods for treating Non-Axisymmetric blunt body flows.

Importance of Landing Site Analysis

The reasons for performing landing site analyses are indicated on the facing page. Landing site access will be the requirements driver for Mars Excursion Vehicle aerobrake L/D and descent profile design.

Importance of Landing Site Analysis

- Landing site location determines the L/D requirements to get from orbit to the site

- L/D requirements determine the configuration of the Mars Excursion Vehicle Aerobrake

- The configuration of the aerobrake determines the load points, vehicle stress and available wake cone area to place the lander vehicle inside of

- The wake cone , stress points,and load points determine the configuration of the lander

- The size and shape of the aerobrake can determine the amount of packaging required for ETO launch and the number of launches required if it is assembled in space

Preliminary Mars Landing Sites Between ± 20° Latitude

The next three pages show a sampling of landing sites of scientific interest in the ± 20° latitude band on Mars. Altitudes are also shown, since altitude has a strong effect on landing delta V. We are presently designing for access to any site within this latitude range, at altitudes up to 5 km. with the COSPAR low-desnity atmosphere. This will permit landings up to 8 - 9 km altitude with typical atmosphere densities.

Preliminary Mars Landing Sites Between +/- 20° Latitude

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Tharsis Montes	5°N	100°	9 km	Ascraeus and Pavonis Mons, rill formations, Tharsis Tholus, unnamed crater
	10°N	82°	3-2 km	Tharsis Tholus, Echus Chasma, Fesenkov Crater, head of Kasei Vallis, Lunae Planum (colored soil)
	-10° S	137°	4 km	Mangala Vallis, Memnonia and Sirenum Fossae, edge of Tharsis Montes shield, Aganippe Fossa, Arsia Mons Colored sands
	-15° S	115°	8-9 km	Arsia Mons, Noctis Labyrinthus, Syria Planum, Claritas Fossae, crater area
Sinai Planum	-18°S	76°	1-8 km	Melas Chasma of Valles Marineras (possible access to Valles Marineras floor); Felis, Melas and Solis Dorsii, crater (unnamed) with rills/flows, Lassell Crater, Coprates Catena

Preliminary Mars Landing Sites Between +/- 20° Latitude page 2

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
South of Eos Chasma	-19°S	49°	3 km	Eos Chasma (part of Valles Marineras, with possible access to the valley floor, accessible places in the valley floor -1 and -2 km) Lassell and Ritchey Craters, Felis Dorsa, crater field some with flow fields, Holden Crater
Hesperia Planum	-16°S	253°	4 km	Tyrrhena Patera (massive flow field from a single source), crater fields, surface cracks and fissures, Terra Tyrrhena area, small mounts
Elysium-Amazonis	0°	180°	0 km	Pettit Crater, Nicholson Crater, surface cracks, Orcus Patera, Cerberus Rupes, colored soils, old craters, Apollinaris Patera, Gusev Crater and flow field, edge of Elysium flow shield, Medusae Fossae, "new" craters in the Elysium flow shield

Preliminary Mars Landing Sites Between +/- 20° Latitude page 3

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
Amazonis Planitia	15°N	155°	0-3 km	flow area around Olympus Mons, edge of Gordii Dorsum and Eumenides Dorsum formations, crater area east of Pettit Crater
Chryse Planitia	18° N	45°	0-(-1) km	Chryse depression (-3 km), end of Kasei Vallis, Sharonov Crater,Lunae Planum; Nanedi, Shalbatanu , Simud, and Tiu Valles, end of Ares Vallis, craters, colored sands

Preliminary Mars Landing Sites Between +/- 20° Latitude page 4

Place	Planet coordinates lat.	long.	Martian altitude	Areas of Interest (accessible by rover, 1000km out from landing)
North of Ganges Catena	-2°S	68°	1 km	Ophir Chasma (part of Valles Marineras, with possible access to the valley floor), Hebes Chasma, Echus Chasma, Juventae Chasma, Ophir Planum, Lunae Planum, crater field, colored soil
Elysium Planitia	19°N	226°	0 km	Hephaestus Fossae, Elysium Fossae,Elysium Mons, Albor Tholus, Eddie Crater with interior formation, colored sands
	19° N	197.5°	2-3 km	Elysium Mons, Elysium Fossae, Ocrus Patera, Cerberus Rupes, Lockyer Crater, Phlegra Montes, colored sands, craters, old and "new"

Mars Descent Analysis Findings

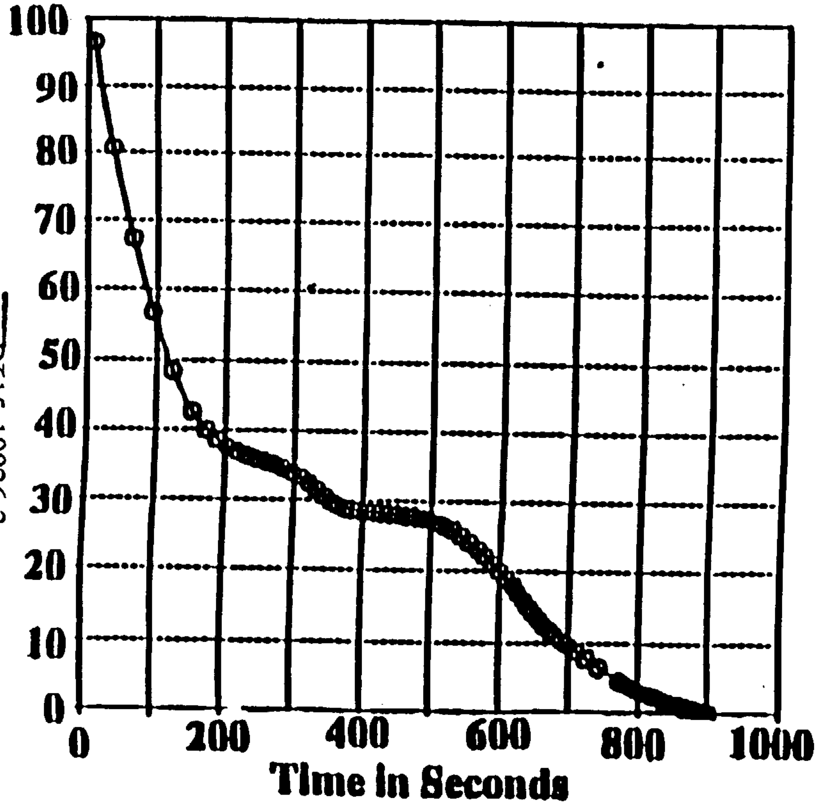
- Wide variation in vehicle crossrange achievable with $0.5 < L/D < 1.0$
- Latitudinal displacement of > 20 deg is achievable for a wide range of entry conditions, for $L/D > 0.9$
- Increasing descent vehicle L/D provides diminishing Delta V requirements
- Landing altitudes of 5 km to 10 km above Mars reference are achievable with present crossrange requirements, for $L/D > 0.95$
- High L/D shape will be a flatter shape then previously investigated with a partial top shield to control wake impingement
- A flare aeromaneuver will be performed at the end of the landing sequence (before vehicle landing for the reusable aerobrake and aerobrake drop for the non-reusable Level II scenario). This flare will be controlled by:
  - For the non-reusable aerobrake, a large flap(.25 of aeroshell area) will separated prior to aerobrake drop.
  - For the reusable aerobrake, the center of gravity control, and therefore the flare maneuver, will be managed by pumping LOX to and from main and auxiliary tanks
  - The reusable areobrake will also have an articulated flap with a maximum area of 10% of the aerobrake

The aerobrake configuration utilized in this analysis is the same as previous. With an L/D of 1.0 the flight time is 2 times greater than for an L/D = 0.5, thereby resulting in range increase of 50%.

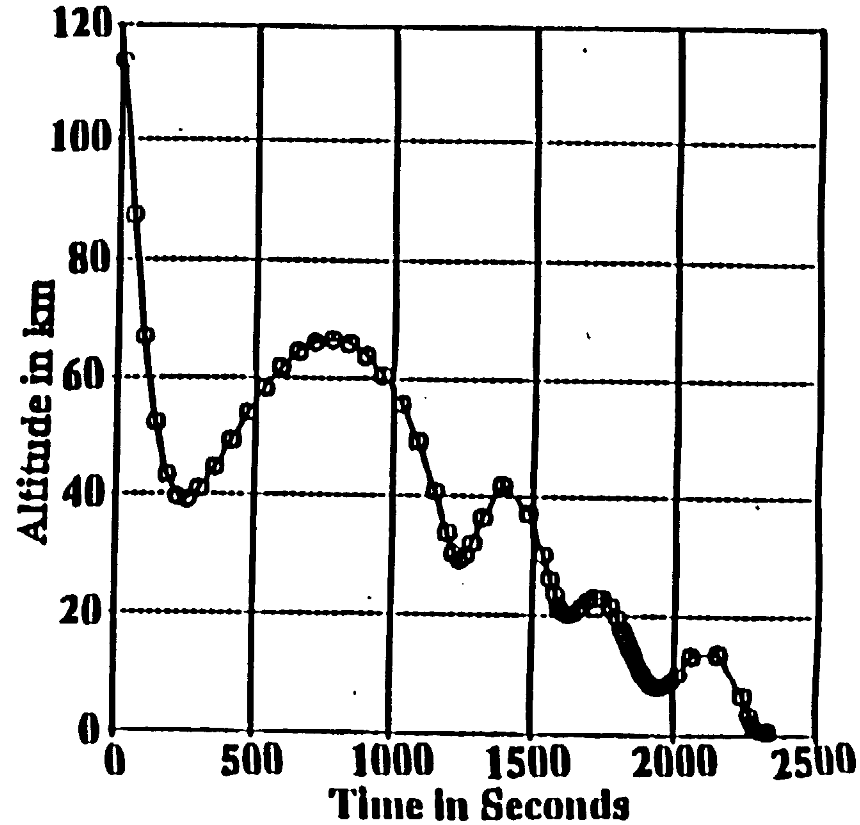
Landing Analysis
Range Effect of L/D

Entry Mass	81 mt
Isp	470 sec
Ref Area	471 m^2

L/D = 1/2



L/D = 1.0



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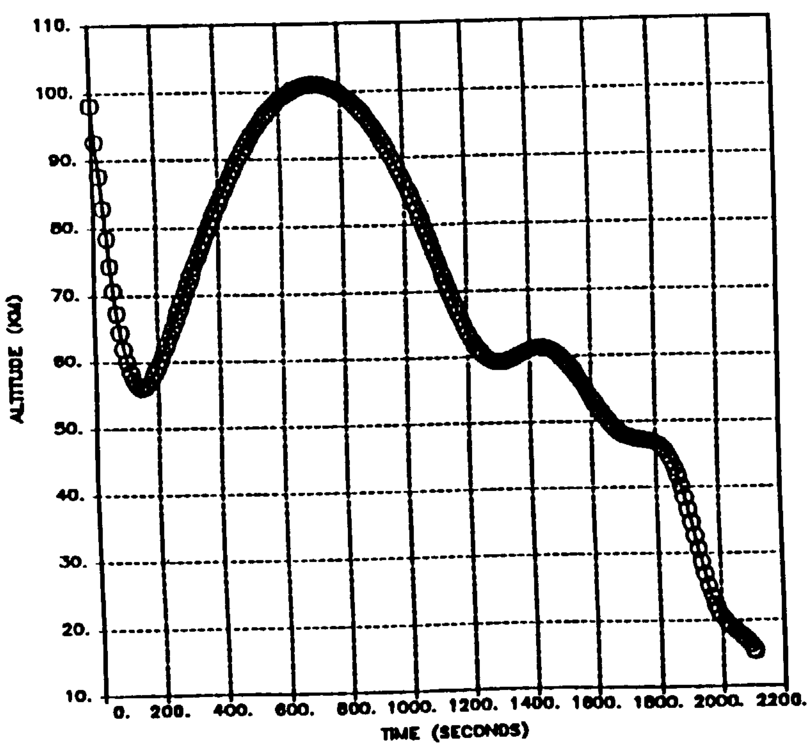
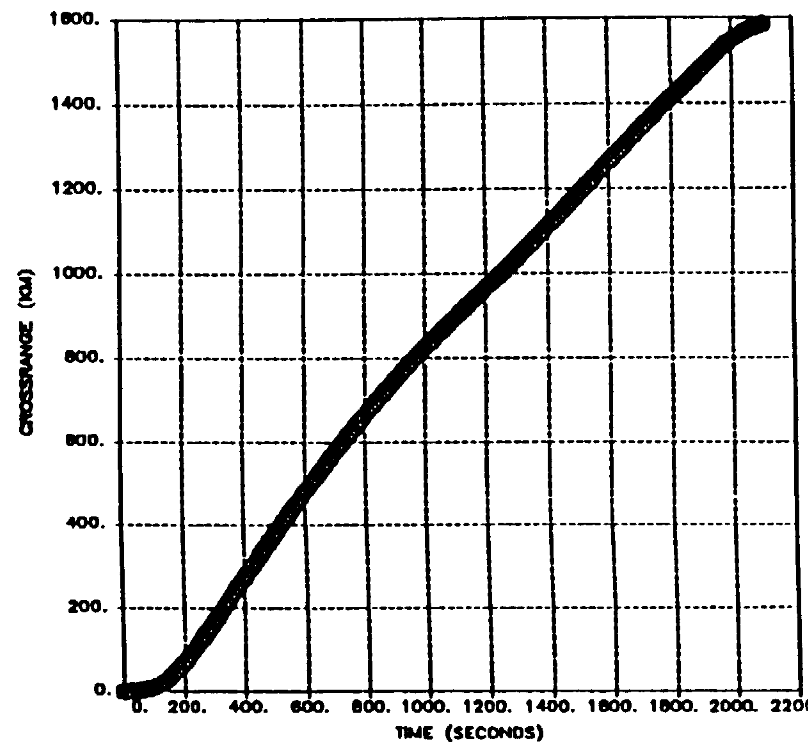
/STCAEM/lev/31May90

The following plots were generated the using OTIS (Optimal Trajectories by Implicit Simulation) program. The plots show the flight path of a vehicle that will give the maximum crossrange and they show the crossrange. The plots were done for a vehicle flying at a lift to drag ratio of one and a lift to drag ratio of one-half.

Lift to Drag Ratio = 1.0





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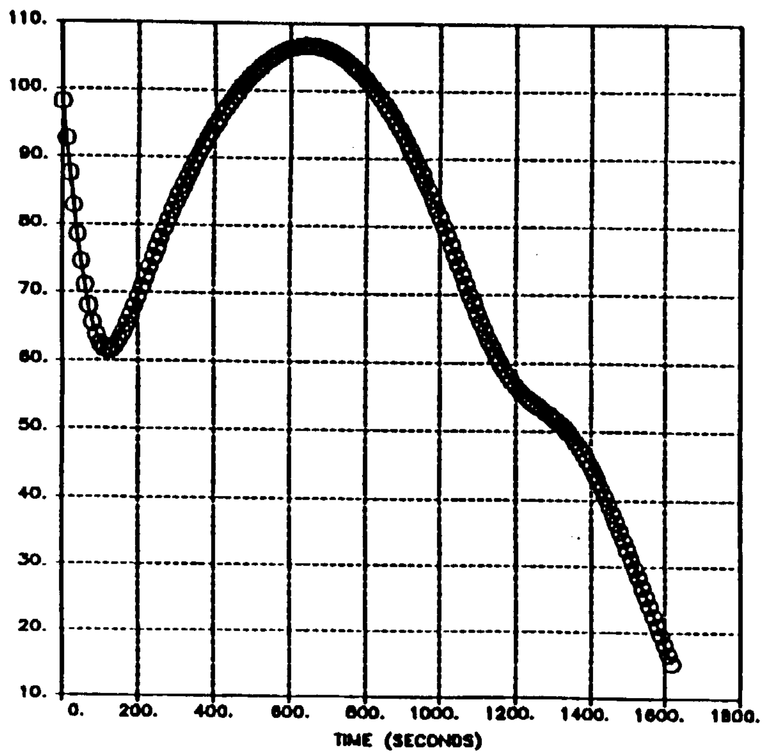
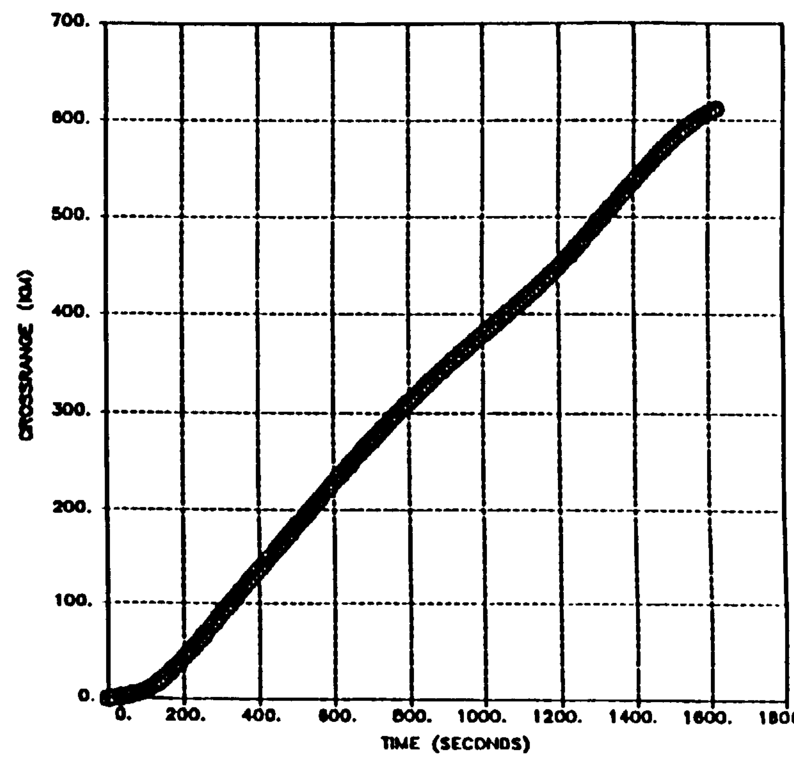
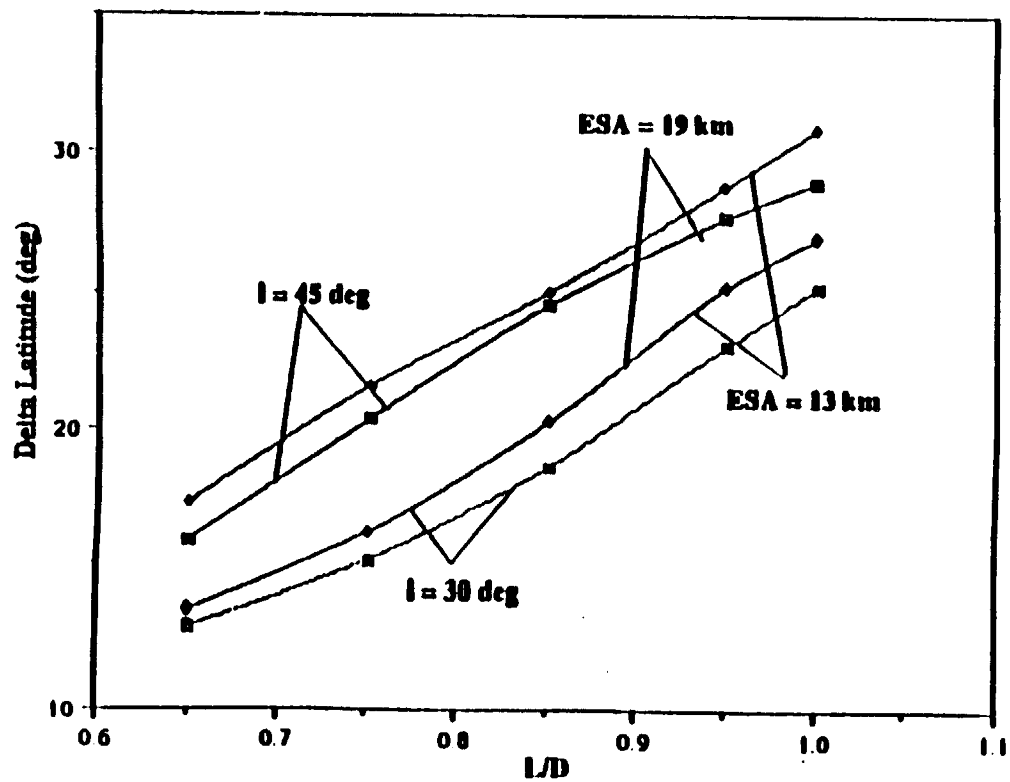
Lift to Drag Ratio = 0.5





The hyperbolic shaped vehicle analysis was carried out for a range of L/D from 0.65 to 1.0. The reference area of the aerobrake was 471 square meters with a total weight of 81 metric tons. The aerobrake weight itself was 9 metric tons. For the analysis, the drag coefficient was held constant at 0.75 and the angle of attack was held at 20°. The calculation was carried out for the cool low density COSPAR atmosphere. For an L/D = 1.0 a change in latitude of 30° is achievable. For the same conditions, a crossrange of 1000 km may be obtained.

Landing Analysis
Delta Latitude



Engine Start Altitude = ESA

Orbital Inclination = I

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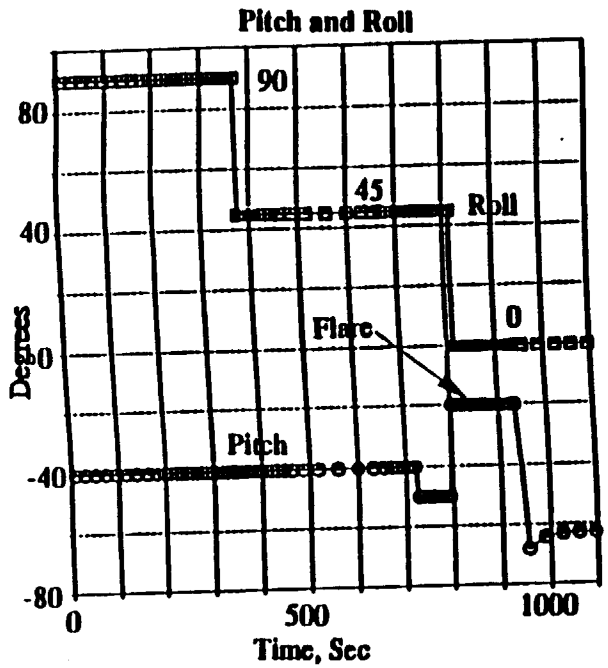
/STCALIM/lev/31May90

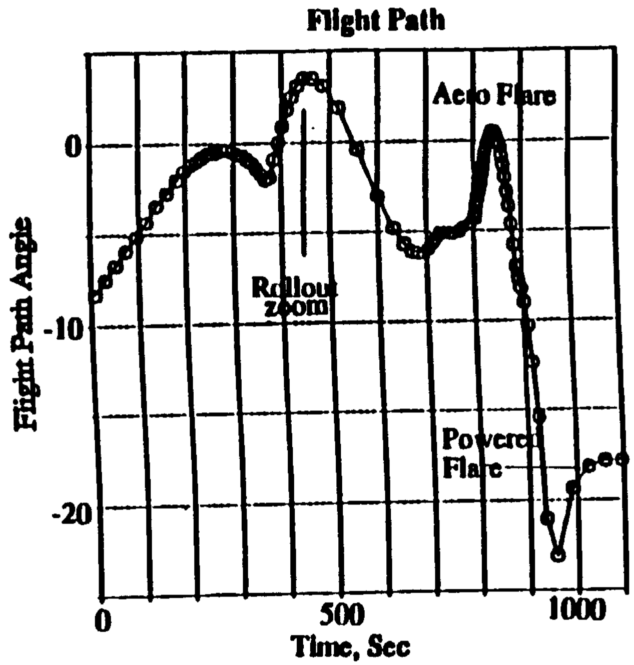
Mars Landing Simulation with Aerodynamic Flare

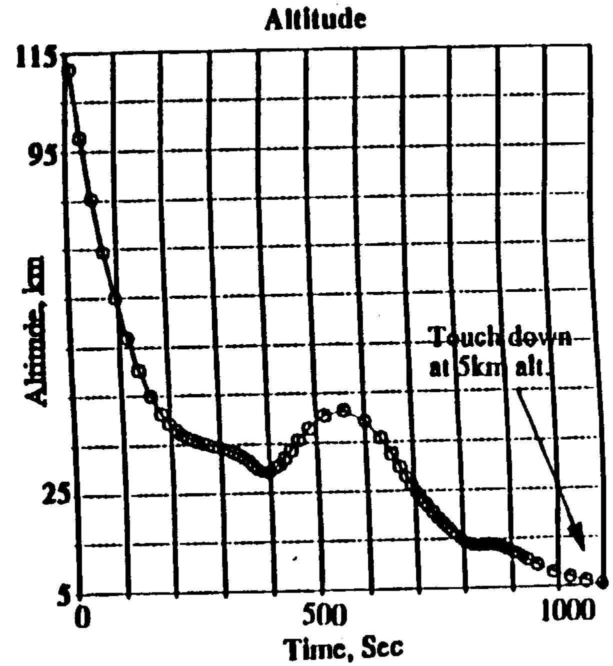
Shown on this chart are results of an un-optimized, but typical, aerodynamic and propulsive descent. Mars entry occurs at a 90° roll angle to obtain maximum cross-range. As the vehicle slows to below circular velocity, roll-out in two steps maintains roughly level flight. Most of the descent is flown at L/D = 1. Prior to engine start, the L/D is briefly increased (drag decreased) to increase speed. Then the vehicle is pitched to maximum lift coefficient at L/D about 0.5. This causes an aerodynamic flare, decreasing speed and increasing path angle. The result is a significant decrease in rocket thrust and delta V for landing.

The importance of this is that it generates a requirement for pitch control, a requirement not present for aerocapture. The combination of high L/D and pitch control will lead to selection of an aerobrake shape much different from the MTV aerocapture case.

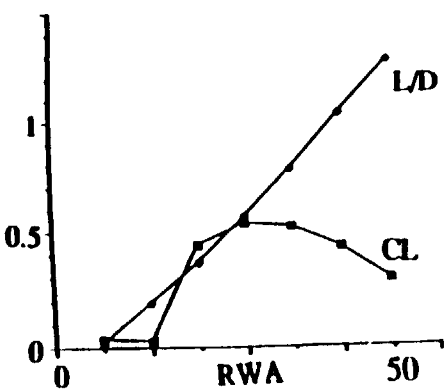
Mars Landing Simulation With Aerodynamic Flare







- COSPAR low-density atmosphere
- Entry mass 81t.
- Thrust 80k
- Max L/D 1.1
- Ref Area 750m^2



For a range of L/D from 0.65 to 1.0, the ideal delta velocity range is from 1400 m/s to 1200 m/s for landing at a 5 km altitude. In the case of an aeroflare with two different reference areas of 470 and 750 m^2 with a cutoff velocity 0.02 the ideal delta velocity is approximately 900m per second. This results in a delta velocity reduction due to using the aeroflare from 200-300 m/s for L/D = 1.0.

Ideal Delta Velocity

Reference Area = 471 sq. m	Reference = 750 sq. m
Thrust = 420 kn	Thrust = 420 kn
ESA = 11.4 km	ESA = 11 km
Aerobrake Drop = 6.79 km	Aerobrake Drop = 8.52
Cutoff Velocity = .1 km/s	Cutoff Velocity = 0.02
Ideal Del Velocity = 529 m/s	Ideal Del Velocity = 918.5

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Levied Requirements

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During the course of the Space Transfer Concepts and Analysis for Exploration Missions contract (STCAEM), Boeing's Advanced Civil Space Systems group (ACSS) has conducted regular review meetings in order to define and derive requirements, conditions and assumptions for systems currently being developed.

As system definition and development progresses, technical experts provide documentation and rationale for requirements that have been derived. This real-time capturing prevents requirements and their associated rationale from being lost or neglected. For example, a vehicle configurator may see the need for providing a minimum passage dimension for vehicle egress or ingress. This requirement would then be captured at an early development stage and would provide a history for the decision. This seemingly simple requirement may have large impacts on the design down the road and its traceability is important.

Derived requirements and rationale are later transfered to the Madison Research Corporation (MRC) where they are then entered into the system data base which has been developed for ACSS using ACIUS's 4th Dimension® software. The data base allows for easy access and traceability of requirements.

The charts that are contained within this document represent two collated copies of principal requirements and assumptions for February 2, and May 30, 1990. The systems defined include: (1) the Mars Transfer Vehicle (MTV), (2) Mars Excursion Vehicle (MEV), (3) Trans-Mars Injection Stage (TMIS), and the Earth Crew Capture Vehicle (ECCV). Each system is then broken down into subsystem headings of: (1) design integration, (2) guidance, navigation and control (GN&C), (3) electrical power, (4) man systems, (5) structure and mechanisms, (6) propulsion, (7) ECLSS, (8) and command and data handling (C&DH). The initials of each of the technical experts responsible for developing the supporting rationale for each of the requirements is indicated parenthetically next to each entry.

Although the majority of the derived requirements listed are directly applicable to all vehicles such as those powered by Nuclear Electric propulsion (NEP), Nuclear Thermal Rockets (NTR), Solar Electric propulsion (SEP) and reference Cryo, there are some that are not. Those requirements that are only directly applicable to a specific vehicle type are indicated within the entry. The italicized entries indicate a modification to an original requirement prior to the second revision of May 30, 1990.

Defining and re-examination of derived requirements will continue through the current contract.

Level II Requirements for Mars Space Transportation

General Requirements

- First Mars landing in 2016
- Cargo flight and second manned landing in 2018
- Vehicles sized to meet given mission phase ΔV budget and durations
- Manned flights to deliver crew and 25t payload to surface
- Crew and 1t payload returned to LEO for each manned flight
- NASA STD 3000 applicable
- Quarantine and medical provisions provided
- Factors of safety set for metallic/nonmetallic/pressure structures
- Failure tolerance/ maintainability requirements identified

Level II Requirements for Mars Space Transportation
(continued)

Mars Transfer Vehicle (MTV)

- **Aerobraking to be used at Mars arrival (Ve $\leq$ 9500 m/sec)**
- **Aerobraking to be used at Earth return (Ve $\leq$ 12500 m/sec)**
- **Provide direct entry capability at Earth return (Ve $\leq$ 14000m/sec)**
- **Piloted MTV to be reusable without major maintenance for 5 missions**
- **EVA capability provided**
- **Zero-G transit**
- **In transit science performed outbound and inbound**

Mars Excursion Vehicle

- **Expendable Vehicle**
- **Chemical propulsion (LOX/LH2)**

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Derived Requirements

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MTV Derived Requirements

- Design Integration
  - Two (2) communications satellites deployed in Mars orbit with total mass = 3000kg (GW)
  - Crew module must accommodate alternative advanced propulsion options (BD)
- GN&C
  - Capture trajectory entry interface for aerocapture not to exceed 6'g' limit and to preclude an uncontrolled skip-out (PB)
- Electrical Power
  - Solar power to be used for transfer phase, batteries to be utilized for sun occultation time while in Mars orbit (BC)
- Man Systems
  - Added protection to crew from Solar Proton Events (SPE) will incorporate use of a "storm shelter". (MA)
  - Consumables stored will suffice for crew residence time from 443-1018 days (includes abort), assumes 100% ECLSS closure of water and oxygen, 0% closure on food and .25 kg leakage per day (PB)
  - Two (2) astronauts able to pass through major circulation paths while wearing EVA suits. (SC)
  - Crew quarters shall provide sufficient volume for casual conversation between at least two (2) crew members (SC)

MTV Derived Requirements

(continued)

- **Man Systems** (continued)
  - Crew visibility during all maneuvers (docking/rendezvous) (SC)
  - There shall be 2 means of egress from each module for emergency escape (SC)
  - Crew module to accommodate 0'g' and induced 'g' environments (SC)

- **Structure and Mechanisms**
  - Airborne support equipment for aerobrake shall be 20% of aerobrake mass (PB)

MEV Derived Requirements

- Design Integration
  - Provide 15% of active weight for spares (JM)
  - MAV must be able to abort-to-orbit during descent phase (PB)
  - Twenty-five (25) ton down payload on manned vehicles (BS)
  - Protective covers provided for all mission critical systems (BS)

- GN&C
  - L/D range from 0.5 to 1.0 (GW)
  - Deorbit from 1 sol x 250 km periapsis orbit (GW)
  - Currently, cross range = ± 500km (GW)
  - Engine start before aerobrake drop (GW)
  - Approach path angle = 15° (GW)
  - Capture trajectory entry interface for MEV aerocapture at Mars not to exceed 6'g' limit on crew members and equipment and to preclude an uncontrolled skipout of the Mars atmosphere (PB)
  - Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1km cep and with beacon assuming 30m cep (PB)
  - Autonomous aerocapture capability at Mars, ~one (1) day before MTV (BS)
  - Aerobrake jettisoned in controlled manner during powered descent phase (BS)

MEV Derived Requirements

(Continued)

- **Propulsion**
  - Pre-descent checkout of engines to be provided (checkout extent TBD) (BD)
  - One (1) meter clearance established between engine bells and surface (SC)

- **Electrical Power**
  - Solar arrays to supply power to MEV following separation from MTV for fifty (50) day approach to Mars (BC)
  - Power for 50 day approach sequence to Mars shall be provided by solar arrays separate from the full MTV configuration. Arrays to be retracted 12 hours prior to Mars encounter, power shall be provided by batteries or other internal source (BC)

- **ECLSS**
  - Capability of two (2) crew cab represses (BD)

- **Man Systems**
  - Consumable s will suffice for a crew residence time of 30 - 600 days dependent on mission stay time and abort scenarios, assumes 100% ECLSS closure of water and oxygen, 0% closure of the food and .15 kg leakage per day (PB)
  - The maximum surface stay time is 600 days (PH)

MEV Derived Requirements

(Continued)

- **Structure and Mechanisms**
  - Shall be at least two (2) functionally independently pressurized areas for emergency conditions The shall be two (2) EVA suits stored in these areas (PB)
  - Establish 30cm clearance between all elements to allow for movement during high-stress maneuvers (SC)
  - Crew cab to have SSF diameter (4.4m), width (1.4m), and penetrations and attachments occur at rings. (SC)
  - Surface hab system to be: removable later by surface construction transport vehicle and protected from damage by MAV blast during ascent start (BS)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Flexible to support reference missions (interconnect design to support reference mission requirements (GW)
  - Fully modularized to utilize ETO capacity , the amount of modularization shall be a function of the ETO vehicle chosen (PB)
  - Assembly to be accomplished on-orbit, remotely and robotically (BS)

- **Propulsion**
  - Reference vehicle is launched "wet" with top-off (dry/wet issue to be traded) (JM)

- **Structure and Mechanisms**
  - Thrust structure - tanks - intertanks used as primary structure (GW)
  - The airborne support equipment mass for launch to Earth is assumed to be 7% for all hardware sets (PB)

Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide (BS)
  - Equipment design life must account for mission duration plus one year (BS)
  - All components designed for 5 missions with refurbishment (except aerobrake) (BS)
  - Design for range of crew sizes, from 4 to 12 (BS)
  - L/D range from 0.5 to 1.0 for aerobrake vehicles at Mars (BS)
- **GN&C**
  - 8500 m/s maximum entry velocity at Mars (GW)
  - 100 m/s error-correction (post aerocapture) (GW)
- **Propulsion**
  - Engine out capabilities in all mission phases (BD)
  - Engine must continuously track C.G. of vehicle from beginning to end of all burns (BD)
  - Maximum gimbal angle of engines TBD (BD)
- **Man Systems**
  - Solar Proton Event (SPE) protection to be provided (MA)
  - Allow for direct viewing of all docking, berthing and landing procedures (SC)

Mars Transfer System Derived Requirements
(Continued)

- **Structure and Mechanisms**
  - All critical function lines and redundant systems shall run non-parallel (PB)
  - All systems shall function up to 2 years in a dormant state and having been subjected to the harsh space environment (PB)
  - The airborne support equipment mass for launch to Earth orbit shall be assumed to be 15% for all hardware except the aerobrake (PB)
  - Airborne support equipment mass assumption for the aerobrake shall be 20% of the aerobrake mass (PB)
  - Aerobrake will be launched to Earth orbit in sections for on-orbit assembly as the reference case (PB)
  - MTV and MEV aerobrakes have common layout of attach points (BS)
  - Vehicle elements will have removable debris shield panel cladding for protection during LEO operations. These panels will be removed and saved in LEO to be used for the next mission-opportunity. The panels will not add to the LEO debris environment (BS)
  - Mission vehicles will carry a robotic manipulation capability to inspect and maintain all exterior areas and systems (BS)
  - Structure optimized to minimize weight, operations, complexity and development effort (BS)
  - Greater than 30cm separation between all major vehicle exterior systems (i.e., tanks, modules) (BS)

- **C&DH**
  - Connectability between links maintained 90% of the time. Availability when scheduled - 98% connectability (PH)

MTV - ECCV Derived Requirements

- GN&C
  - Capture trajectory entry interface for ECCV aerocapture or aeroentry into Earth atmosphere not to exceed 6'g' limit on crew and personnel, and to preclude an uncontrolled skip out of Earth atmosphere (PB)
  - L/D = 0.25 (MF)
- Structure and Mechanisms
  - Interior materials must conform to NASA standards for outgassing, fire hazards,etc. (SC)

MTV - TMIS Derived Requirements

- **Design Integration**
  - Assembly to be minimized to extent practical. (KS)
- **Propulsion**
  - Passive thermal control system including zero-'g' thermodynamic vent system coupled to multiple vapor cooled shields. (JM)
  - TMIS insulating system is a continuously purged MLI over foam design optimized for minimum ground-hold, launch, and orbital boiloff. Includes vapor cooled shield (coupled to TVS) outside of foam. (JM)
  - TMIS tanks launched late in assembly sequence to minimize orbital stay time before TMI burn (, 6 months). (JM)
  - MTV tank insulation system is thick (2-4") MLI blankets. Multiple vapor cooled shields placed at optimum points in the MLI. (JM)
- **Structure and Mechanisms**
  - *Thrust structure - tanks - intertanks used as primary structure for cryo/aerobrake only* (GW)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

Mars Transfer System Derived Requirements

- **Design Integration**
  - Wake closure cone behind all aerobrakes is 44° wide. The total wake closure angle is centered on the velocity vector. (BS)

- **GN&C**
  - *200 m/s error correction (post aerocapture)* (GW)

- **Propulsion**
  - *Engine out capabilities in all mission phases. NTR engine out capabilities TBD* (BD)
  - All passive cryogenic thermal control system.
  - No. MTV-TMIS fluid transfer before Earth departure. (MEV tanks refrigerated or filled after MOI)

- **Structure and Mechanisms**
  - Aerobrake externally mounted to vehicle for launch to Earth orbit ("Ninja Turtle" concept) (PB)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MEV Derived Requirements

- **Design Integration**
  - Down payload on manned vehicles
    - ~ 25 mt down payload for reference MEV (includes habitat module) (BD)
    - ~ 0.7 mt down payload for the 'Mini-MEV' (crew habitat is provided by the ascent/descent cab) (BD)
- **GN&C**
  - *Currently, cross range = ±1000 km for high L/D aerobrake* (GW)
  - *Landing approach path angle = 15°* (GW)
  - *Landing accuracy after aerobrake jettison will be unaided by landing beacons assuming 1 km CEP and with beacon assuming 30 m CEP* (PB)
- **Propulsion**
  - Engine out capabilities for ascent/descent stages (BD)
  - Passive cryogenic storage system: MLI with vapor cooled shields (JM)
  - Gravity field environment eliminates need for zero-'g' acquisition and venting. (JM)
  - Vacuum jacketed ascent tanks for Mars boiloff reduction. (JM)
  - MEV propellant transferred from MTV prior to descent. (JM)
- **Electrical Power**
  - *Solar arrays to supply power following separation from MTV for ~ 50 day approach to Mars. Arrays to be retracted TBD hrs. prior to Mars descent (cryo/aerobrake).* (BC)
  - Batteries or fuel cells to provide power for ascent and descent phases. (BC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

MTV Derived Requirements

- **GN&C**
  - *Capture trajectory entry interface for aerocapture options not to exceed 6'g' limit and to preclude an uncontrolled skip-out* (MC)
  - Aerocapture exit errors not to exceed 0.25° inclination, RAAN, and ARCP, and a 0.1 hr period (MC)
  - GN&C requirement for advanced propulsion TBD: (MC)
    - ~ NTR - capture into planned orbit ± TBD
    - ~ EP (electric propulsion options) - TBD

- **Electrical Power**
  - *Solar power to be used for transfer phase, batteries or fuel cells to be utilized for sun occultation time while in Mars orbit except for NEP.* (BC)
  - NEP power derived from existing power system with a backup energy supply via fuel cells (BC)

- **Man Systems**
  - Volume per crew guidelines extrapolated from historical data (SC)
    - ~ Transfer hab = 112 m^3/crew
  - Two independant pressurized volumes for safety (SC)
  - Gravity condition emphasized to accommodate 0-'g' and 1-'g' and for surface commonality (SC)
  - 2.3 m standard ceiling height for psychological and locomotion (SC)

- **Structure and Mechanisms**
  - All penetrations occur in barrel section to minimize mass. (SC)

Note: Changes to existing derived requirements dated 02 February 1990 are shown here in italics

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Guidelines and Assumptions

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Mars Initiative - Assumptions

- Multi-impulse TMI and TEI is permitted, (engine restart) (e.g., three-burn departures acceptable for TMI to ease launch declination window problems [Level II])
- Cryogenic propulsion for Earth/Mars departures and Mars descent (cryogenic/aerobrake for Earth and Mars are selected as reference)
- Proven cryogenic storage technologies will be used
- Advanced propulsion technology options include NTR. SEP, NEP, and GCR
- MTV expendable on "difficult" opposition missions; return to Earth via ECCV
- TMIS expendable for reference system
- 100 ton cargo requirement (cargo mission) met by two (2) standard MEV's without ascent stages
- Maximum size surface payloads on piloted MEV: 6 m diameter and 13 m length

Contingency, Flexibility and Reserves

Flight Performance Requirements and Reserves

- 2% ΔV for Space Transfer Vehicles
- Add 2% for performance requirements uncertainties in selected instances
- Compute finite burn losses and add to impulsive requirements.
- Include delta V requirements for launch windows from LEO.

Dry Mass Contingency Allowances

- None for existing hardware
- None for consumables and impulse propellant
  - Consumables requirements shall include needed mission flexibility allowances.
  - Propellant reserves generated by flight performance reserves.
  - Use 2% of tank capacity for liquid and vapor unusable propellant; counts as inert mass.

- 5% on slightly modified hardware
- 15% on new design/known technology
- 15%-25% on new design/new technology, complex design, and poorly-understood requirements.

Payloads include flight support equipment

Manager's Reserve Policy, e.g. between launch vehicle capability and manifesting, TBD.

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III. Operating Modes and Options

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Reference

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Cryo/Aerobrake - Operating Modes and Options Reference

This section contains the following:

- Operations Outline
- Operations Task Flow Description
- Operations Assumptions
- Operational Task Flow

In order to evaluate the difficulty of the mission operations a top level view of the necessary sequence of events was generated. Only the areas of on-orbit assembly and ground support were delved into to any depth. These are discussed in the support section of this document. The path itself is shown in this section and includes options at assembly (on or off Space Station), in transit outbound or inbound (with or without Venus flyby, Deep Space Burn and coast correction any combination of which may be used) and on return (depending on how much of the vehicle is recovered and where it is recovered at).

The Cryo/ Aerobrake vehicle will operate out of the LEO Space Station orbit. The completed vehicle will leave from a position co-orbiting with the Space Station and do one to three burns to attain the Declination Launch Asymptote (DLA) required for Mars transfer. The Trans-Mars - Injection stage is dropped after the last burn and the transit configuration established. This, at present, is the zero-gravity transit configuration, but an artificial gravity configuration would be established at this point in flight. For any swingby, Deep Space Burn or coast correction maneuver, the artificial gravity configuration must be despun, reconfigured to the zero-g conditions and reconfigured to the artificial gravity conditions after the maneuver has been performed.

The on-line self-check capability of the systems and subsystems will be used throughout the mission to monitor the vehicle health and indicate preventative maintenance. Due to the length of the mission (1-3 years) the vehicle must be self sufficient and capable of maintenance and repair with a limited crew (4-7 people). The length of mission time and the distance will impose limits on the communications and control of the vehicle that can be done by ground operations; the crew are on their own resources.

About 50 days prior to Mars entry the Mars Excursion Vehicle (MEV) and the Mars Transit Vehicle (MTV) will separate, with the MEV operating autonomously and entering first as a pathfinder, the two vehicle sections will aerocapture and rendezvous in orbit. If anything happens to the MEV in capture, the MTV with crew, will abort and return to Earth. After the vehicle sections are docked and the final site selection has been made, the MTV will be set to operate autonomously, the crew will transfer to the MEV, demate the MTV and MEV, perform the on orbit checkout and descend to the surface. The MEV will have the capability to perform a descent abort with the ascent section in the event of an emergency to obtain orbit. From there, a rendezvous and docking maneuver with the MTV will be done for crew transfer and Earth return.

Once on the surface, the MEV establishes contact with both the automated MTV and Earth., then proceeds to carry out the surface mission. When the surface mission is complete, the ascent section liftoff leaving the descent section of the lander and surface habitat behind. The ascent section attains orbit and docks with the MTV, the crew transfers with the return samples and all extraneous mass is jettisoned prior to the Trans-Earth -Injection Burn.

The inbound return transit proceeds like the outbound leg, with options in Venus swingby , coast maneuvers and transit flight configuration. On Earth return, the baseline option is to have the crew and samples transfer to the Earth Crew Capsule Vehicle (ECCV) several days before Earth entry takes place, disengage from the MTV and return to Earth on a direct entry course in the style of the

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Apollo crew capsule. Alternate capture scenarios involve capturing the ECCV into a Space Station access orbit and crew return through the Space Station, and full capture of the MTV into LEO orbit

Cryo/Aerobrake Mission Operations Outline

This is a top level outline of the major task sequences and their relative location for the Cryogenic fuel/Aerobrake vehicle for Mars missions. It is divided into four segments:

a) Near-Earth Operations (initial)- involving operations from hardware buildup to Trans -Mars Burn

b) Transit Operations - operations to be performed during the outbound transit flight from Trans-Mars Burn through the MTV/MEV separation for Mars atmosphere aerocapture

c) Mars Operations - covers the events from MEV and MTV aerocapture through the Trans -Earth Burn

d)Transit and Near Earth Operations - looks at the inbound transit return to Earth and capture operations at Earth

These segments will be further broken down into distinct top-level tasks in the Mars Operations Task Flow for the Cryo/Aerobrake.

Mars Major Mission Operations
Cryo-Aerobrake

Near- Earth Operations:

- Ground testing and support
- Launch support and Launch
- On orbit assembly and checkout
- Departure positioning and TMI burn

Transit Operations:

- TMI stage drop
- Transit flight configuration
- Optional maneuvers- Venus swingby, Deep Space Burn, coast correction and reconfigure
- Periodic Maintenance and inspection
- Autonomous checks and separation maneuvers

Mars Operations:

- Aerocapture, rendezvous and dock
- Separation
  MEV: Land and establish base, ascend
  MTV : Autonomous orbit, communications relay, survey
- Rendezvous and dock
- Jettison excess mass
- TEI burn

Transit and Near Earth:

- Transit configuration
- Optional Maneuvers- coast correction , Venus Swingby, reconfigure
- Earth Return- ECCV direct entry or ECCV orbit capture or MTV capture

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Mars Mission Operations Task Flow
Cryo/Aerobrake

This is a task by task top-level operations task flow for the Cryo/Aerobrake to complete a general Mars mission. Those tasks that are optional (may or may not be used on any given mission) are shown with darkened background. Those operations flows that present alternatives t o the baselined flow or are alternative actions are shown with dashed flow lines.

The assumptions under which these flows were developed are these: Communications with Earth is periodic, for the most part the decision making capability is with the astronauts beyond the TMI burn. This is due to the long time lag in communications (up to an hour) with Earth. Critical maneuvers such as flybys, Deep Space Burns, craft separations and docking, landing and ascent, and TEI burn will be reported to Earth prior to and after the event. Autonomous system self check capability will be present on all vehicle systems used for assembly and maintenance. Robotic assembly of the vehicle- MTV and MEV done at SSF/ on orbit, TMI assembly and integration, propellant top-off and final inspection and checkout done off-station.
and the manned transfer vehicle is self-suffient in repair capability

Assumptions

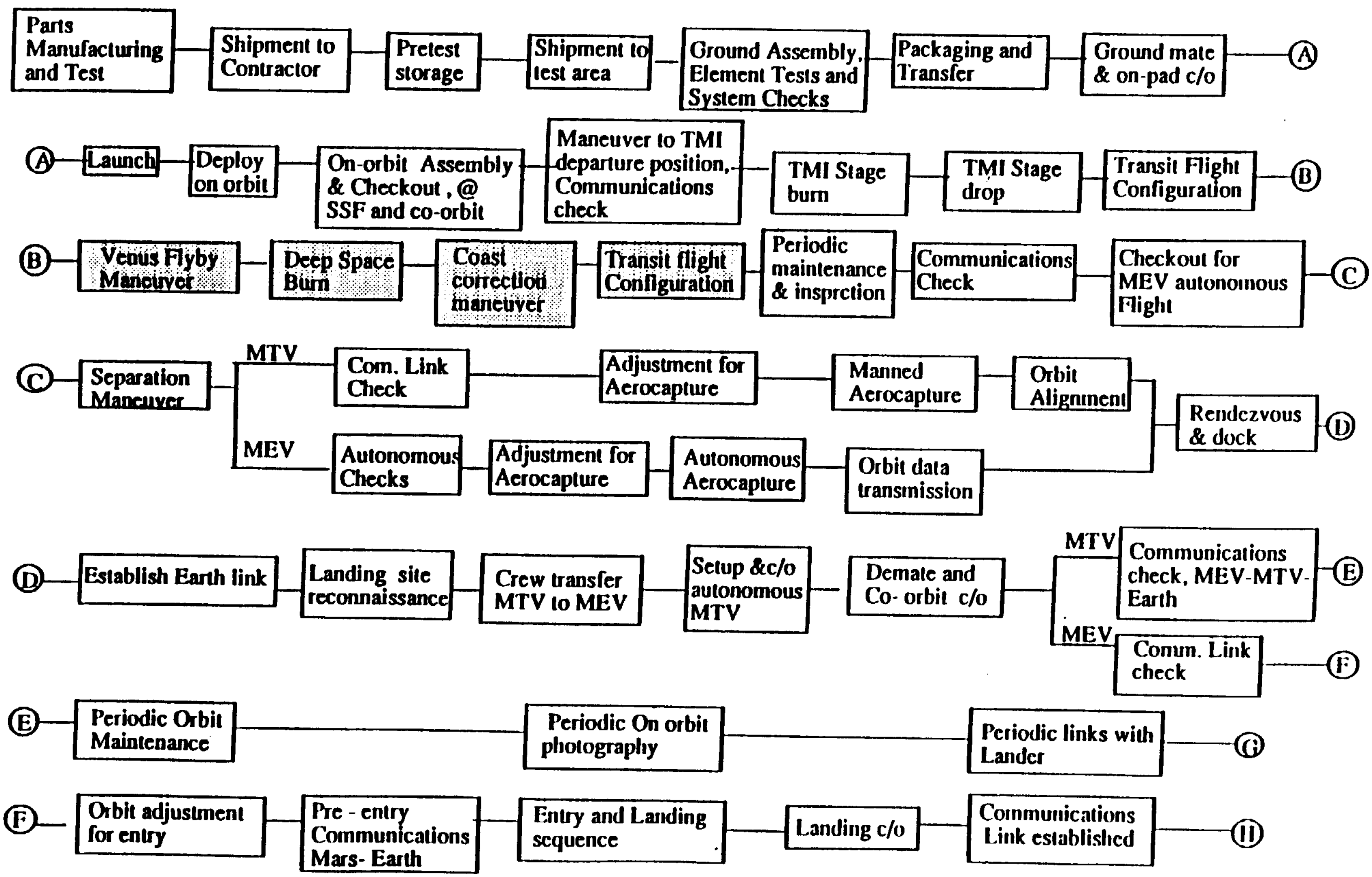
- Communications with Earth is periodic, for the most part the decision making capability is with the astronauts beyond the TMI burn. This is due to the long time lag in communications (up to an hour) with the Earth. Critical maneuvers such as flybys, Deep Space Burns, craft separations and docking , landing and ascent, and TEI burn will be reported to Earth prior to and after the event.

- Autonomous system self check capability present on all vehicle parts used for assembly and maintenance

- Robotic assembly of the vehicle - MTV and MEV done at SSF/ on orbit, TMI assembly and integration , propellant top-off and final inspection and checkout done off- station

- Manned transfer vehicle is self-suffient in repair capability

Mars Mission Operational Task Flow
Cryo/Aerobrake



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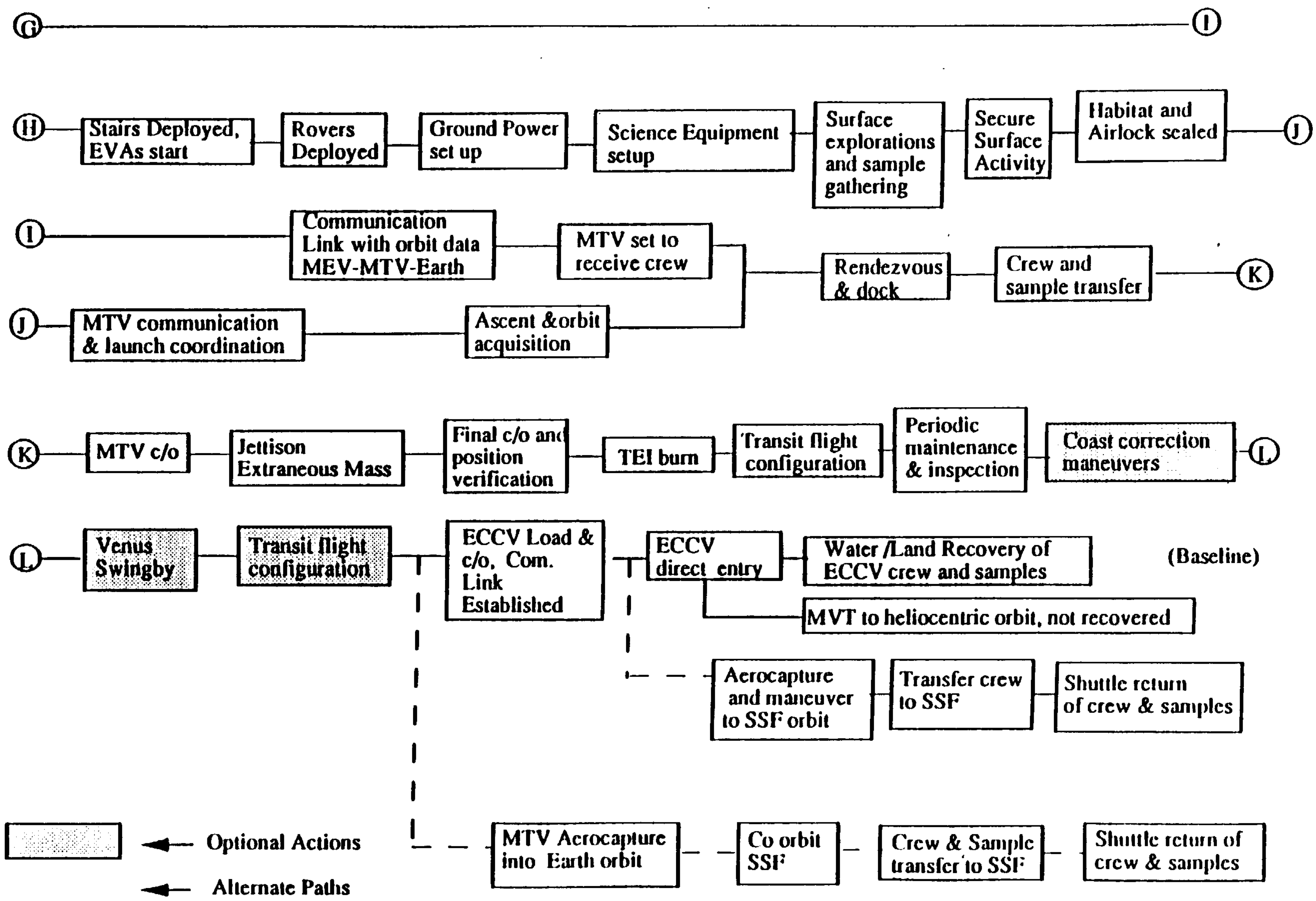

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Other

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Cryo/Aerobrake - Operating Modes and Options - Other

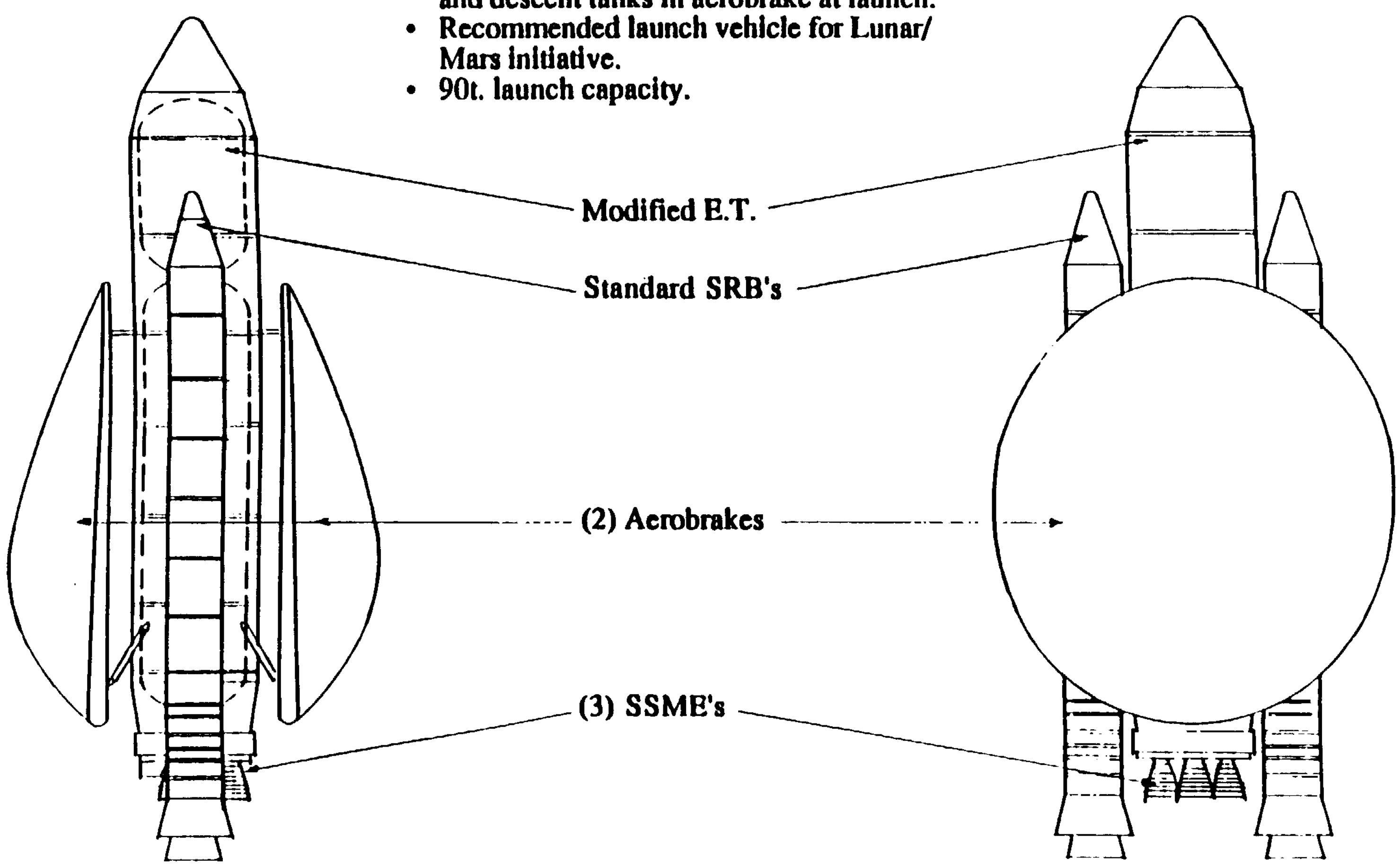
Presented here is a launch configuration for the externally mounted, fully assembled aerobrake. This option has been called "Ninja Turtle" launch configuration. While the initial investigation of launching the vehicle externally mounted indicated that "more work" had to be done the regard this as a viable option; some launch considerations such as shrouds and fairings were not considered in the original calculation which was based on a design sketch. The analyses also involved the launch of two aerobrakes instead of one and was, again, a preliminary analyses. We believe that this launch configuration deserves further analyses. It parallels the configuration of the Shuttle and would solve the problems of on-orbit construction of the aerobrakes and severely volume limited launches

Shuttle Derived Aerobrake Launch Option

In manifesting the Mars vehicle we examined alternatives to launching the aerobrake in sections, which is a volume limited load. The result was this "turtle" or "piggyback" configuration that may be able to take additional payload up secured to the areobrake and/or the inline section. This is still a preliminary design which must be analyzed for launch loads and aerodynamics.

Shuttle Derived Aerobrake Launch Option

- Aerobrake completely assembled on Earth.
- Eliminates in-orbit assembly/check-out.
- Eliminates packing problem.
- Uses shuttle derived launch vehicle.
- Can possibly pack MEV lander structure and descent tanks in aerobrake at launch.
- Recommended launch vehicle for Lunar/ Mars initiative.
- 90t. launch capacity.



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IV. System Description of the Vehicle

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Parts Description

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IV. System Description

A. Part Descriptions

The first set of charts tabulates subsystem characteristics for the seven flight vehicle elements of the cryogenic/aerobrake vehicle. The following chart presents trade item decisions and rationale for subsystem choices.

The Mars excursion vehicle (MEV) is packaged into an asymmetrical aerobrake for Mars capture and landing. The shape of the brake and the configuration of the MEV are driven by the (assumed) 22 degree wake deflection inward of the velocity vector streamline. The other packaging consideration is placing the center of mass in line with the aerodynamic force vector. An alternative configuration includes a Mars surface reconnaissance (MSR) vehicle which is mounted on the aerobrake for Mars capture. The MSR vehicle lands at a different site than the manned lander, then returns a sample to the manned lander for return to Earth.

The Mars transfer vehicle and Mars excursion vehicle are docked together during the planetary transfers to gain the use of the combined volumes for crew habitation. A short duration crew module is used for return to orbit. It carries a crew of four. The MEV includes the crew module, descent and ascent stages, and a surface habitat. The MEV has a landing leg span of almost 20m and a height overall of 14m. Several views are provided from a computer solid model of the MEV.

An Earth crew capture vehicle is used for crew return to the Earth's surface. The configuration shown is for a crew size of five, although subsequent analysis indicates a crew of six is required to provide adequate crew skill redundancy.

Habitation Module Weight Trade Study. This study considered different module shapes for varying crew size, to determine least weight solutions. Primary and some secondary structure were considered in this study as weight discriminators. It was assumed that certain penetration-related secondary structure (airlocks, hatches, windows) and interiors (non-pressure bearing floors and walls, and equipment mounting) would not be significantly different in weight across the options. The results indicate larger diameter (i.e. more spherical) modules are lighter than the SSF design for large crew sizes.

The selected module concept has a 7.6m diameter and a 2:1 aspect ratio with elliptical end domes.

Cryogenic Boiloff Code Tank Estimation. Tank characteristics as a function of operating pressure and multilayer insulation thickness were estimated. The estimates generated by the computer model agree well with the actual mass of the Space Shuttle External Tank when the External Tank capacities are used.

Relative Development Effort Comparison. Estimates of the development effort for each propulsion element in a total Lunar/Mars program were made for various combinations of propulsion. The nuclear thermal rocket yielded the lowest effort estimate on a relative scale. This is only a gross comparison, not considering the differing cost of propulsion developments.

Subsystem Summary

The following three charts contain a concise listing of the primary characteristics of seven major subsystems for each of the seven major reference vehicle stages.

ADVANCED CIVIL SPACE SYSTEMS

Subsystem Summary

BOEING

Subsystems	MTV Crew Module	MTV TMI Stage	MTV TEI Stage	MEV Descent Stage	MEV Ascent Stage	MEV Crew Module	ECCV
Structures	7.6m Dia. x 9m L Al, axial tension tie internal bulkhead	Ribbed skin, Thrust struct., LOX/LH2 tanks, plumbing, etc. Prop. fract. = 0.9	Truss tube, Thrust struct., LOX/LH2tanks, plumbing, etc. Frame struct. 5% of TEI prop & inert mass	Thrust struct, tanks, etc. Frame struct 10% of inert mass. Landing legs - 3% landed mass	Thrust struct, tanks with vac shell, etc.	4.4m D 0.5 ellipsoidal Al shell	3.9m x 2.7m Apollo type capsule
Thermal Control	Water/Glycol w/body mounted radiator integral w/meteoroid shield	Passive	Passive	Passive	Passive	Water/Glycol w/ ext. panel radiator & water flash evaporator	Water/ Glycol
Aerobrake structures	Rigid, Deep shell assembled in segments, Spar ribbed: Honeycomb face sheets. 13% cap mass.	N/A	N/A	Same dimensions as MTV aerobrake 13% of capture mass	N/A	N/A	N/A
Thermal	Reradiative TPS	N/A	N/A	Reradiative	N/A	N/A	Ablative Shield

Subsystem Summary (continued)

Subsystems	MTV Crew Module	MTV TMI Stage	MTV TEI Stage	MEV Descent Stage	MEV Ascent Stage	MEV Crew Module	ECCV
Avionics	SSF derived command, cntrl, & data handling equip., GN&C platforms, new comm. systems, health monit., aerobrake attitude control	Main propulsion controls / instrumentation	Main propulsion C&I solar array positioning control, RMS pos. ctrl. unit, Aeroshell integrity monitor	Main propulsion C&I, Aerobrake attitude control	Main propulsion C&I, Cryo prop monitor sys.,	Apollo/LEM type complete flight ctrl system Onboard health monitoring equip	Apollo Command Module type
Power	15 kW, solar arrays w/ battery storage	Distribution sys.,post separation battery power	Distribution sys., Back-up fuel cells system	Distribution sys., Back-up fuel cells system	Distribution sys. Back-up fuel cells system	2.3 kW fuel cell for Des/Asc, solar arrays for surface	Battery storage
Propulsion Engines	N/A	5-200k lb adv. engines (ASE),w/eng. out cap., stage T/W=0.4, Large area ratio,Isp=475	3-34k lb w/engine out cap. Stage T/W=0.2,Large area ratio, Isp=475	4-34k lb w/engine out cap. Extendible/retrac. nozzles Isp=460	2-34k lb. w/engine out cap. Ext/ret nozzles. Isp=460	N/A	N/A

Subsystem Summary (continued)

Subsystems	MTV Crew Module	MTV TMI Stage	MTV TEI Stage	MEV Descent Stage	MEV Ascent Stage	MEV Crew Module	ECCV
ECLSS	SSF derived with all resupplies and change-out equip. onboard. Closed on H2O and O2	N/A	N/A	N/A	N/A	Open loop Apollo type	Open loop Apollo type
Crew Accomm.	40-50 cubic meters habitat volume/person. Dedicated radiation shelter. SSF level of crew comfort; shower etc.	N/A	N/A	N/A	N/A	54 cubic meters total habitat volume, Spartan crew accom. 3-day nominal occupancy	8 cubic meters Apollo type crew accom. 3 - day nominal occupancy

Mars Mission Trades & Issues

Several trade items or issues are listed with the applicable selections for the reference vehicle given as well as the rationale behind the selection. The two boxed trade items are examined in more detail in subsequent charts.

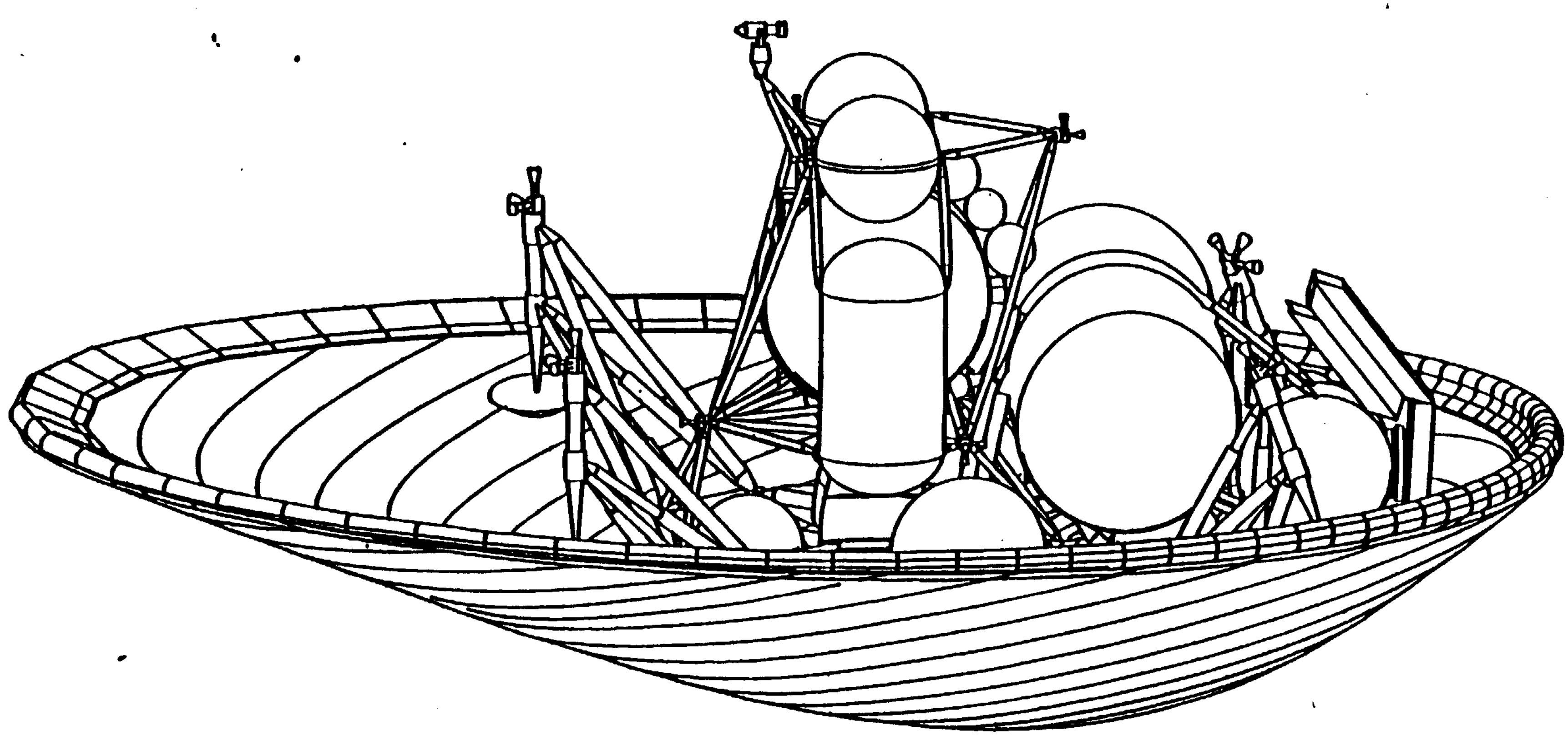
Mars Mission Trades & Issues

Trade item	Selection	Rationale
MEV		
Propulsion:		
Staging	2 stg, aeroshell dropped during desc.	Alternate split stg very attractive
Engine thrust	30-34k lbf,	eng commonality w M dep
Number of Engines	2 asc, 4 desc	min number of eng's w eng out cap.
Current vs ASE	Current design, low Pc, conservative Isp=460	yet to be traded
Propellant	Cryo O2/H2	Min mass even at 600 day surf stay
Thermal insulation	MLI,vapor cooled shields,vac jacket-asc	Maximum passive insulation
Configuration:		
Asc cab	Streched ellipsoid, bi-level	Min mass,ease of egress,SSF derived
Tank placement	O2 top, H02 mideck-asc, desc on frame	Thrust thru CG w eng out
Cab power	Fuel cell,	3 day occupancy
Surf ace power	Solar array, fuel cell,	30 day requirement
TEI		
Propulsion	3 x 30-34k lbf eng's for M dep,	Common w MEV eng's,
Parking orbit	1 sol by 250 km periapsis	Min Delta V
ECLSS	SSF derived, near 100% closure	Proven hardware
Reuse/recovery	Crew return via ECCV, no reuse	Min IMEO
TMI		
Propulsion	5 x 150k lbf Isp=475 engines	Performance, eng delivered w tank set
propellant loading	tanked delivery	yet to be traded
Staging/recovery	Single stg, no recovery	yet to be traded

Mars Excursion Vehicle (MEV)

The MEV packs into the aerobrake as shown. The landing gear fold up, the descent engine nozzles retract, and the surface stairs fold up for packaging. The aerobrake acts as a heat shield during descent; 60 seconds prior to aerobrake separation, the engines extend through doors in the aerobrake and fire. The landing gear deploys, jettisoning the aerobrake before terminal descent.

MEV/Aerobrake Configuration



MEV Aerobraking Constraints

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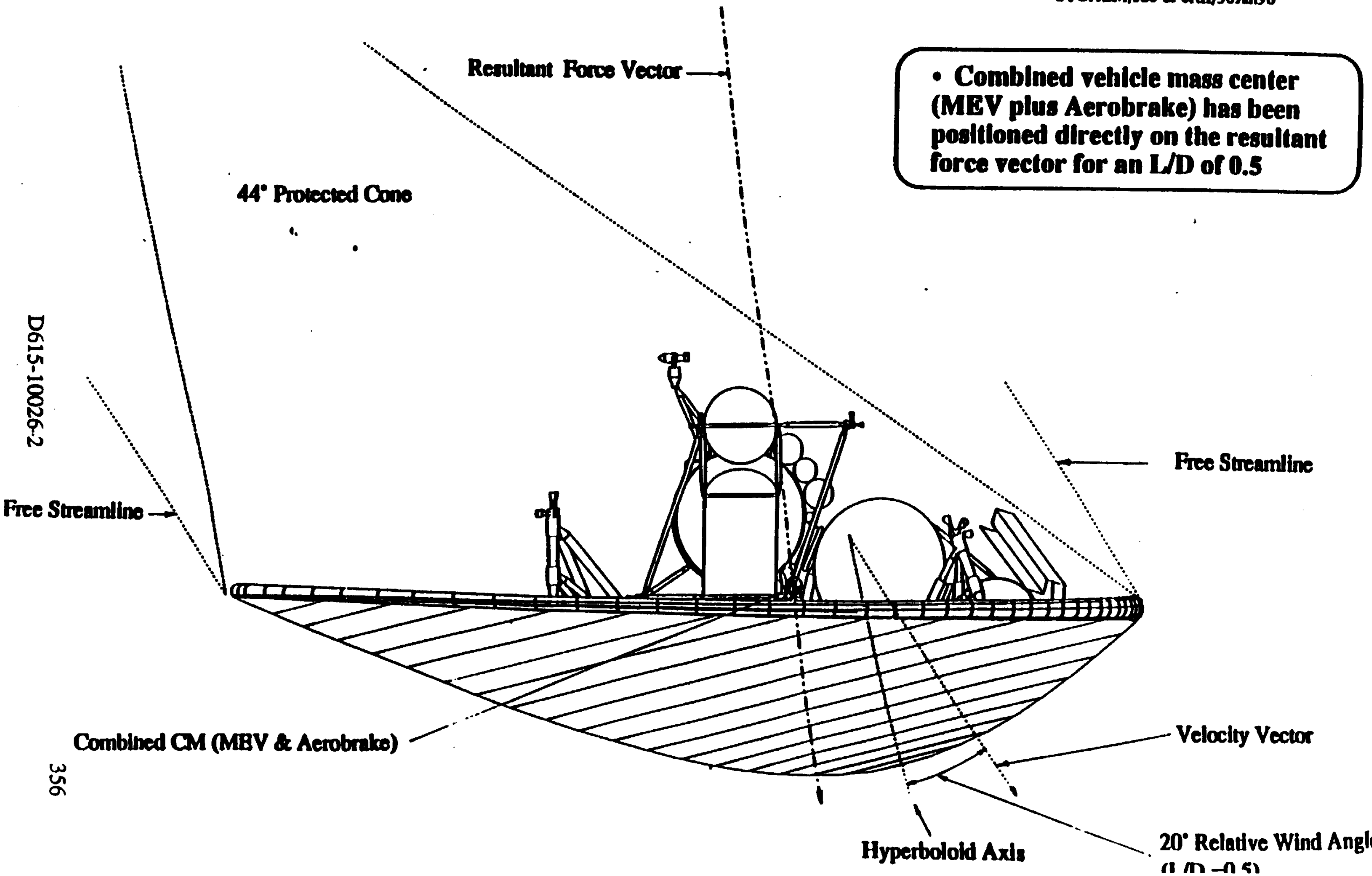
- Combined vehicle mass center (MEV plus Aerobrake) has been positioned directly on the resultant force vector for an L/D of 0.5



MEV with Mars Site Reconnaissance Vehicle

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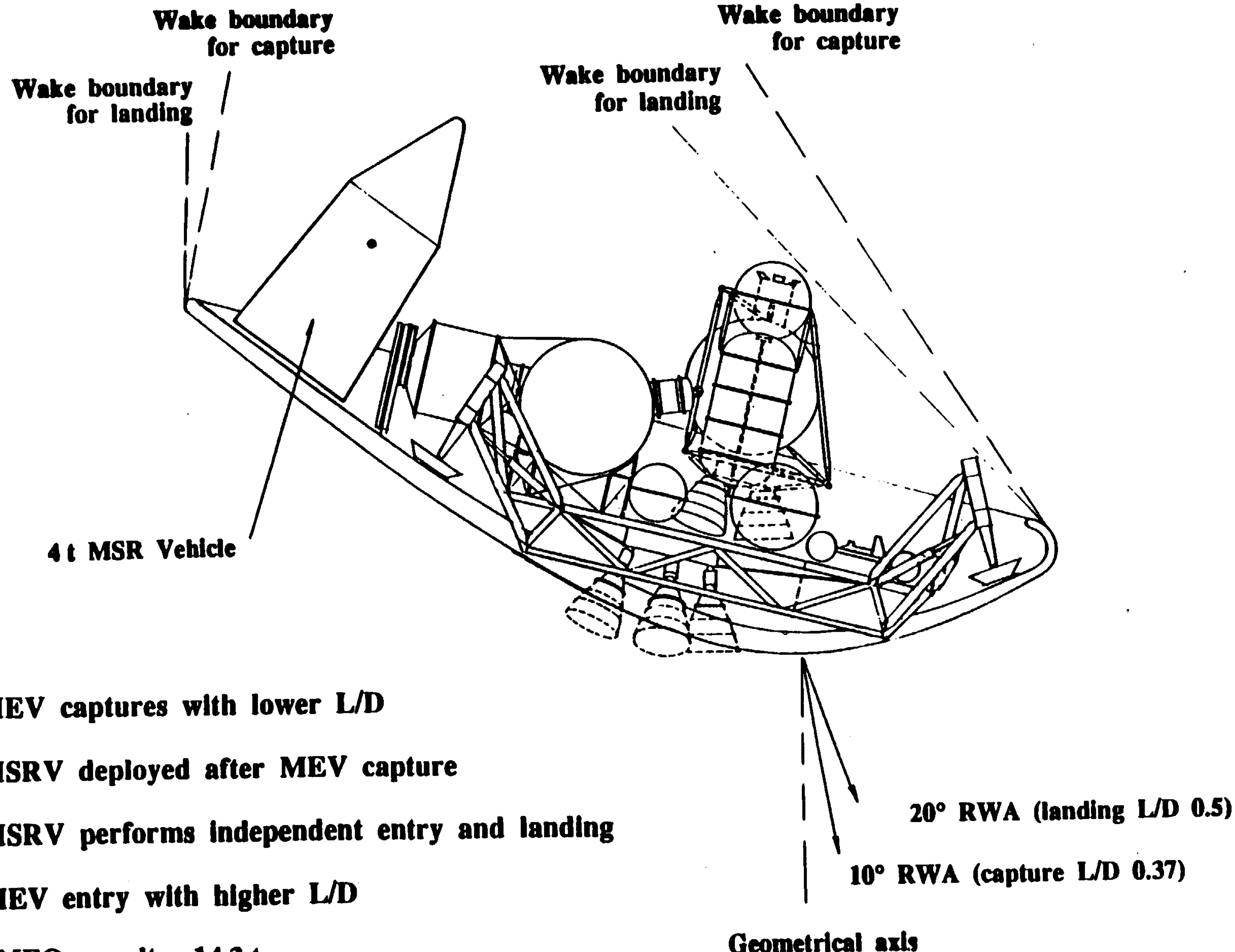
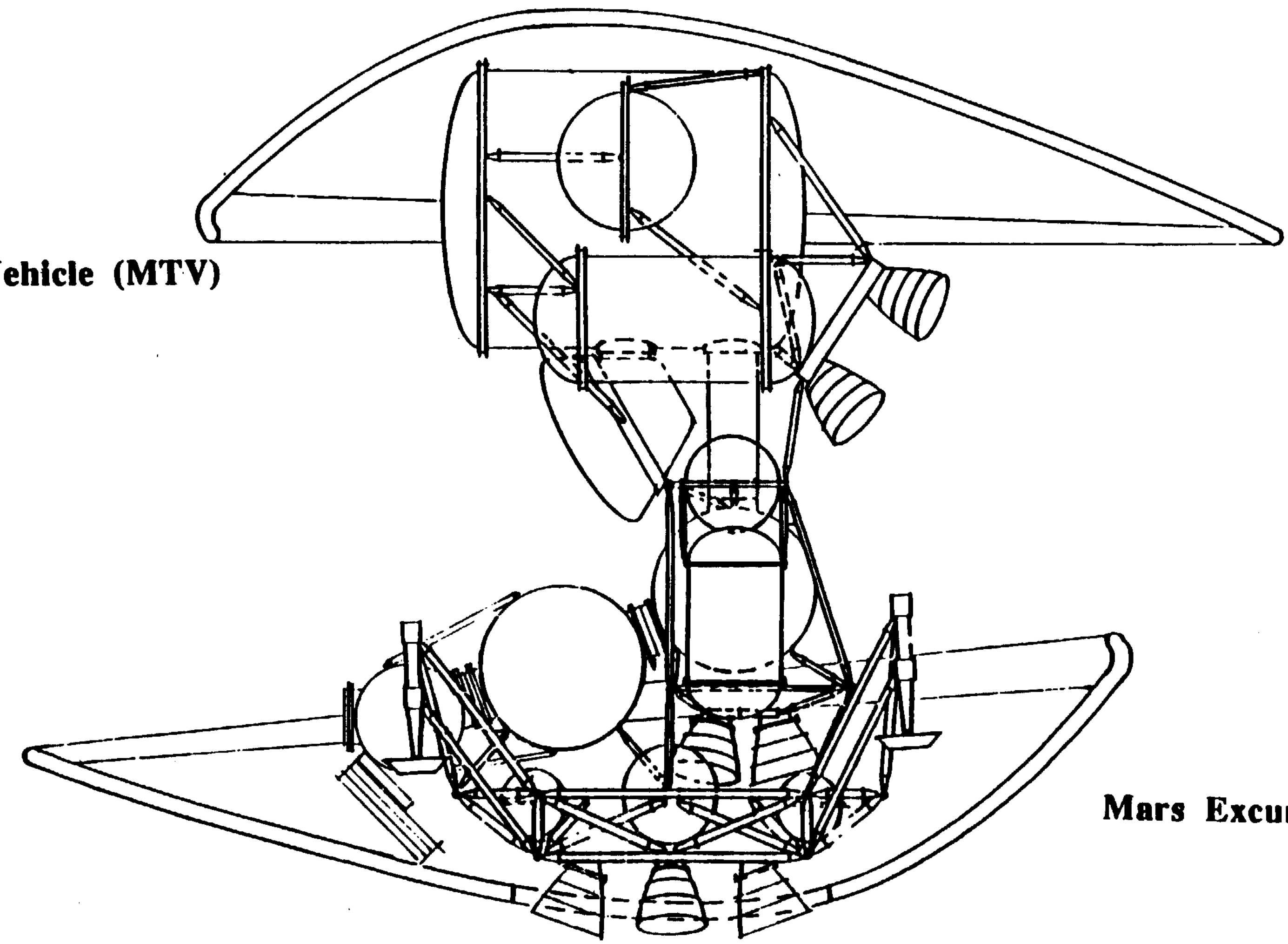
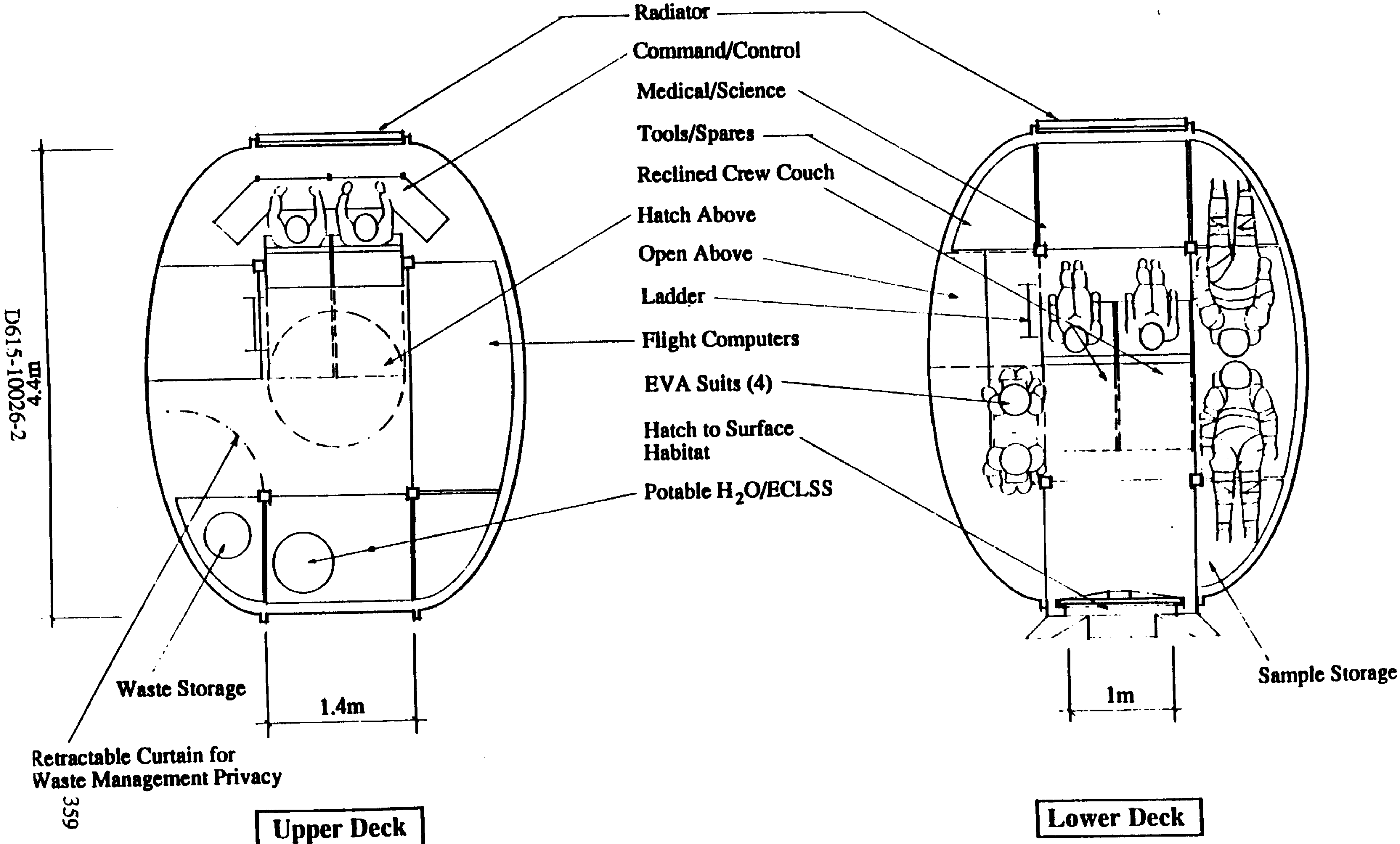


- MEV captures with lower L/D
- MSRV deployed after MEV capture
- MSRV performs independent entry and landing
- MEV entry with higher L/D
- IMEO penalty 14.3 t

Berthed MTV and MEV



Common Short-Duration Crew Module



Common Short-Duration Crew Module

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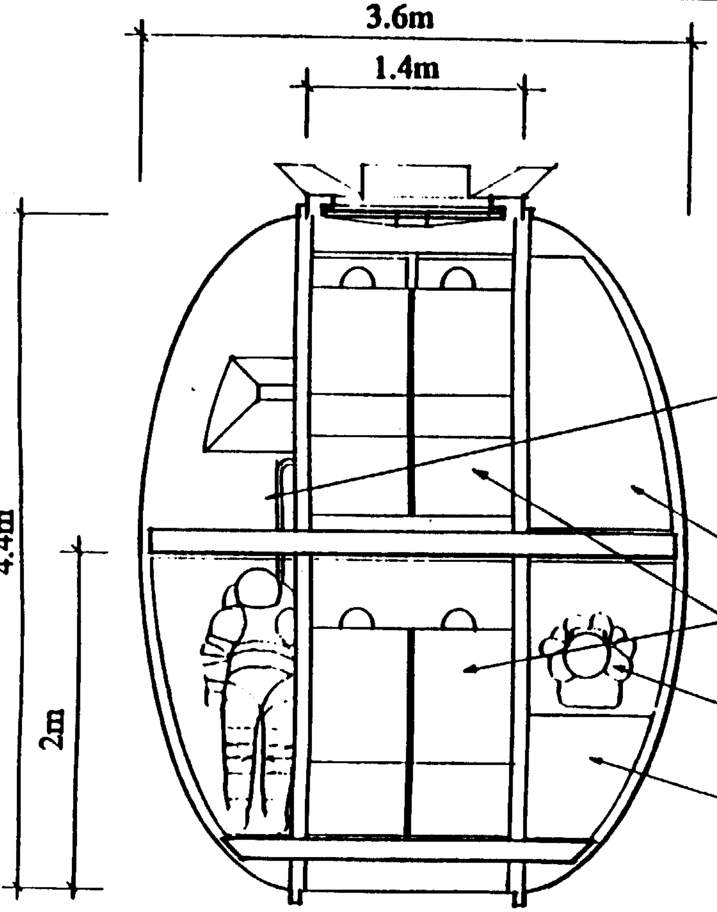

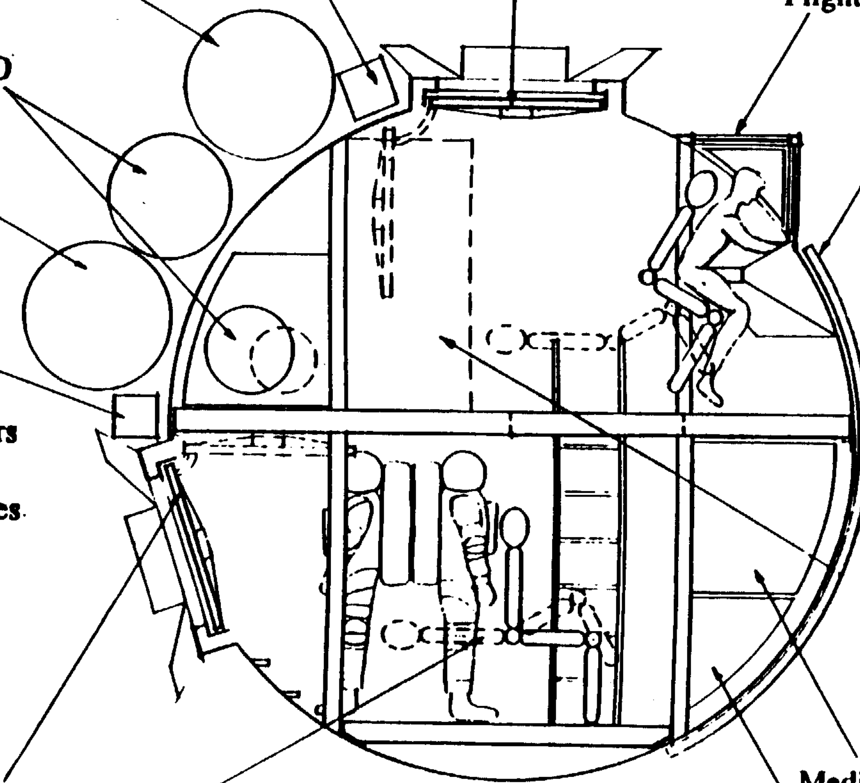

- SSF diameter cylinder.
- All penetrations occur in cylinder section.
- All structural attachments at girth rings.
- Common ellipsoidal end domes.

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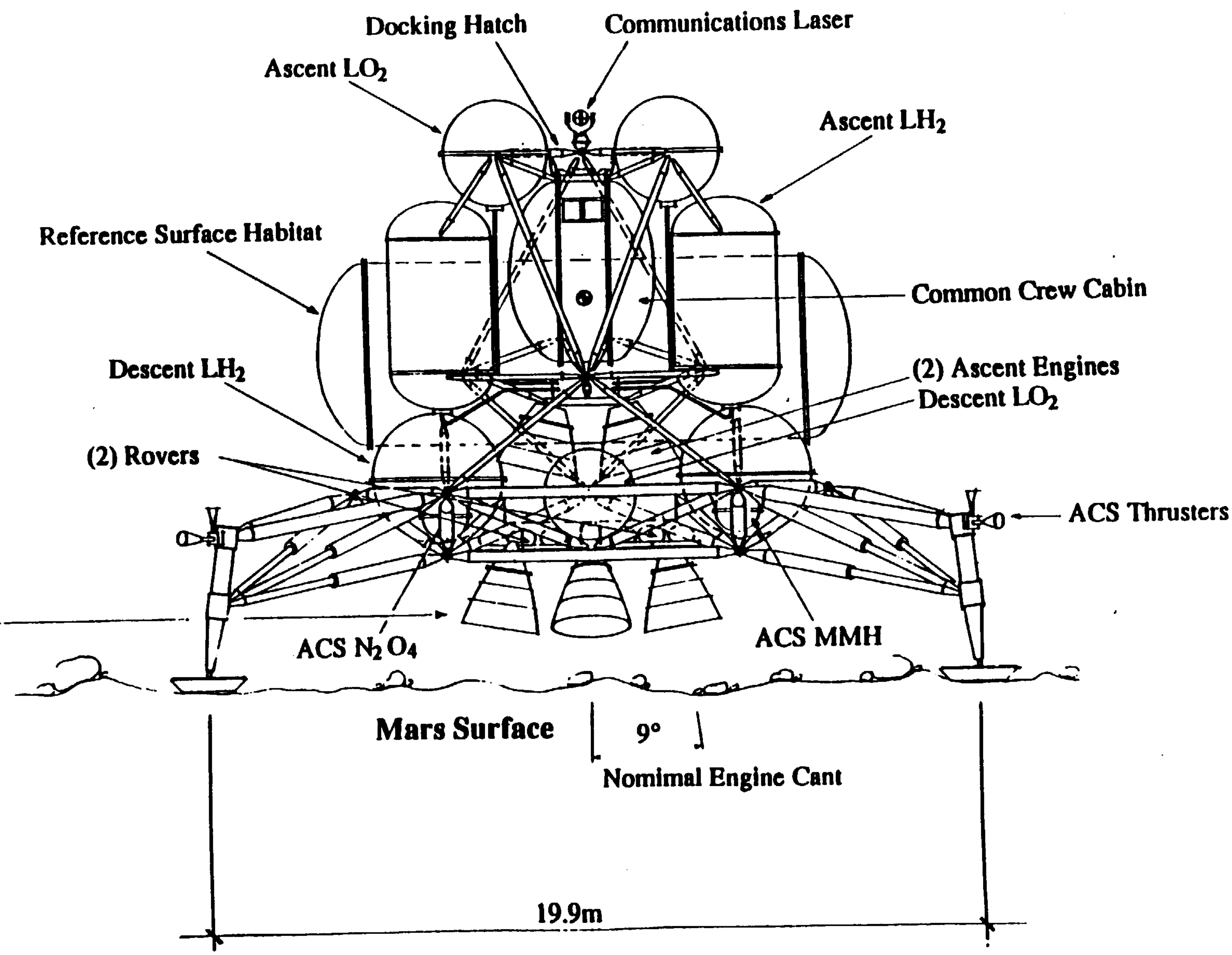
Mars Excursion Vehicle (MEV)



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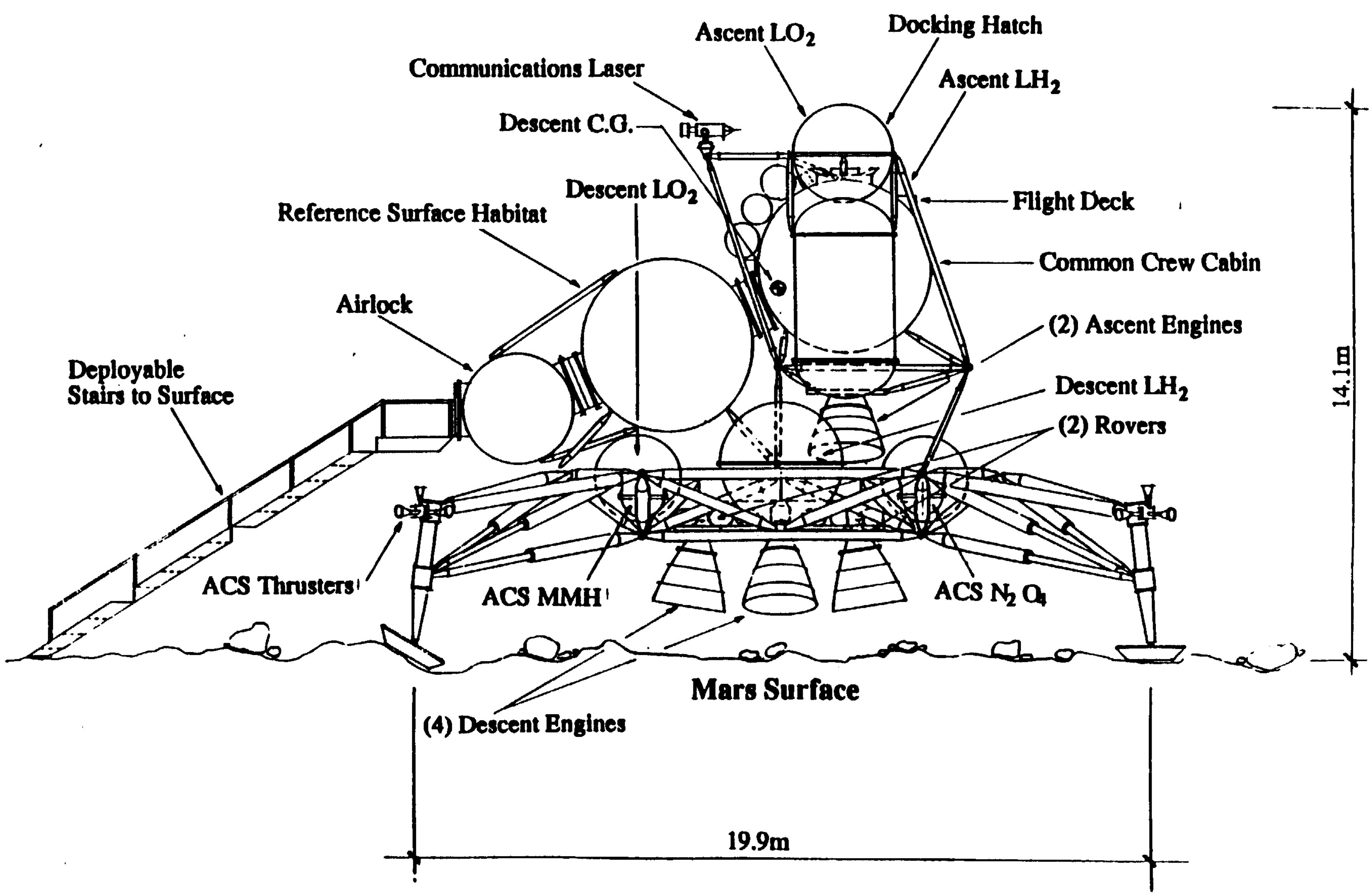
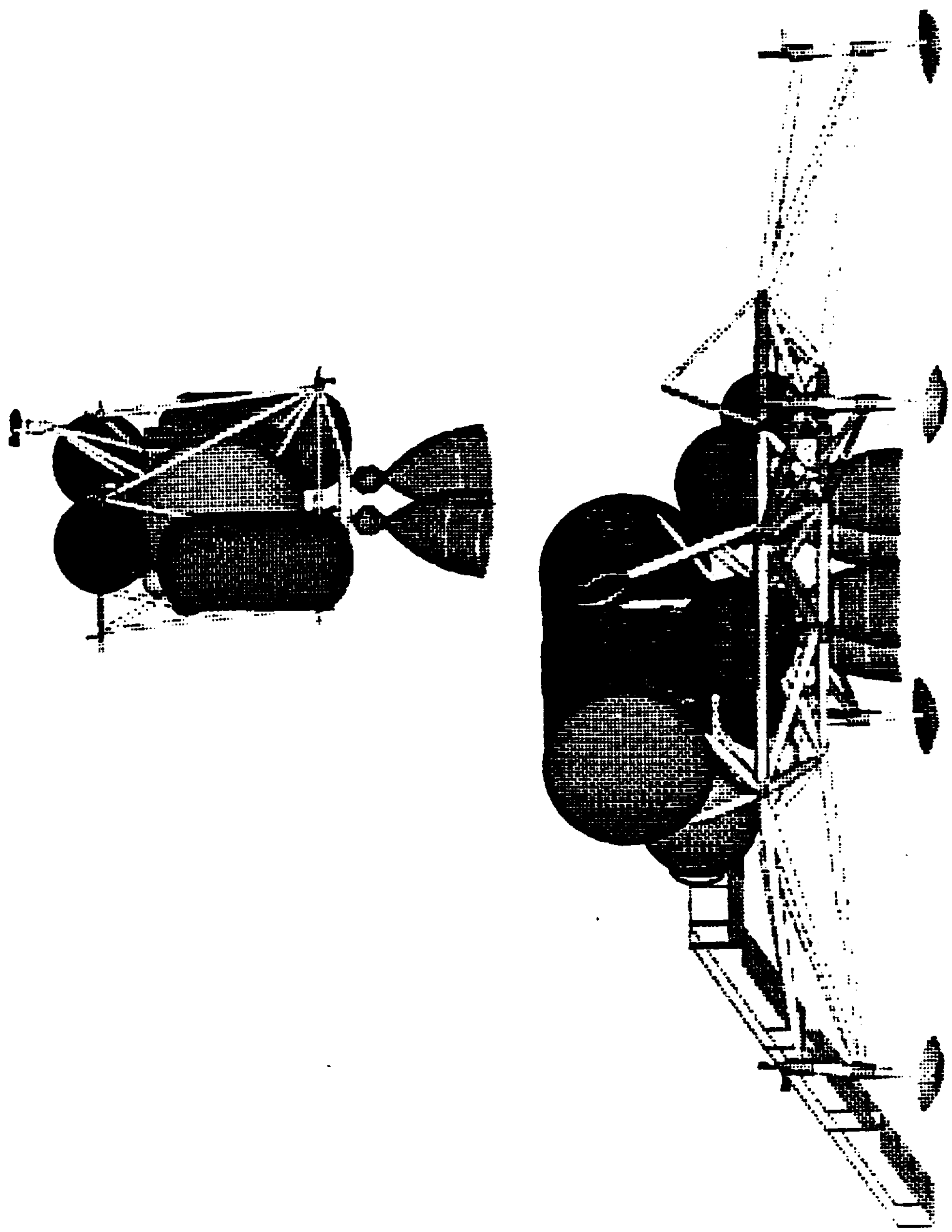
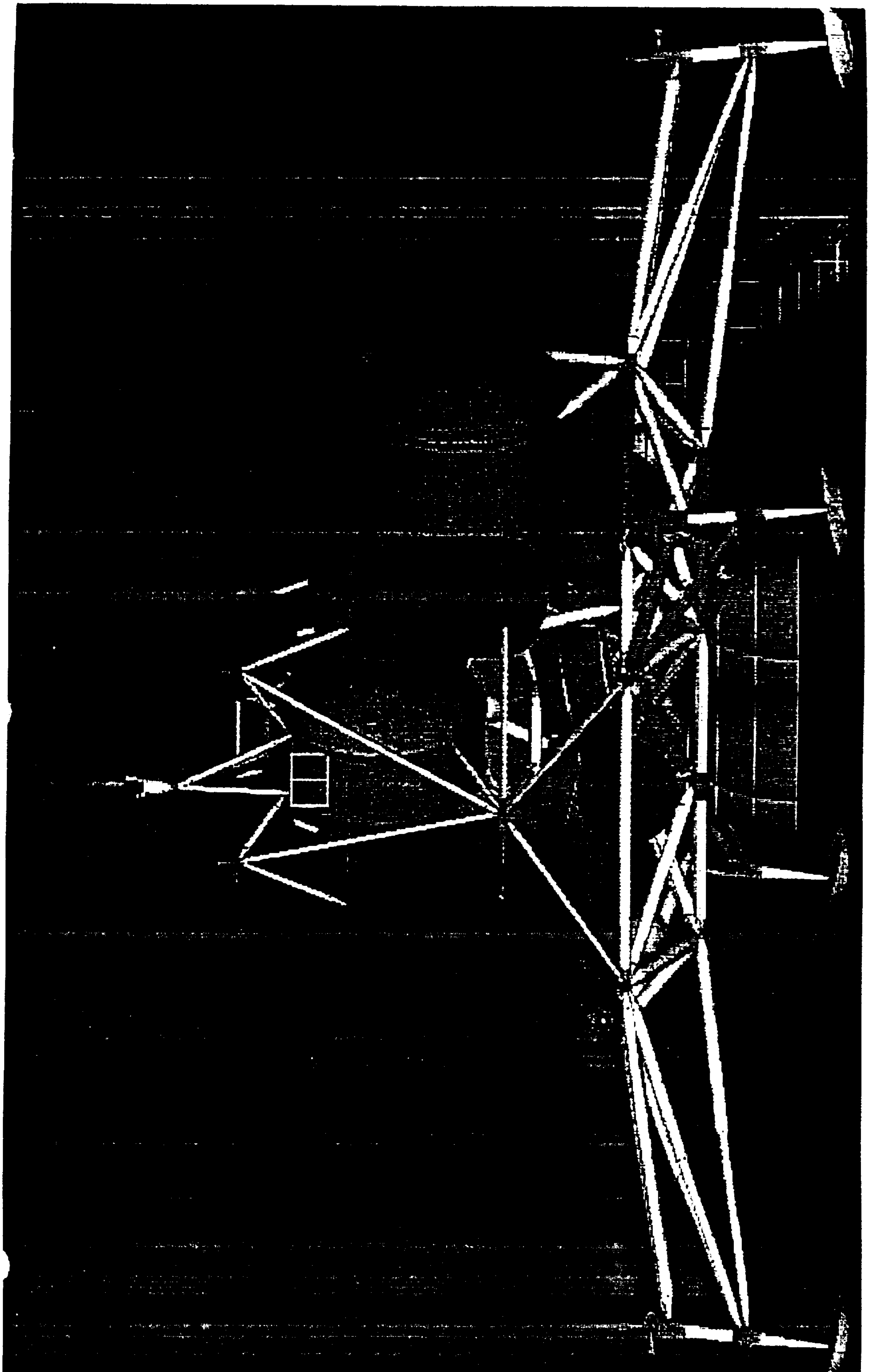
Mars Excursion Vehicle (MEV)







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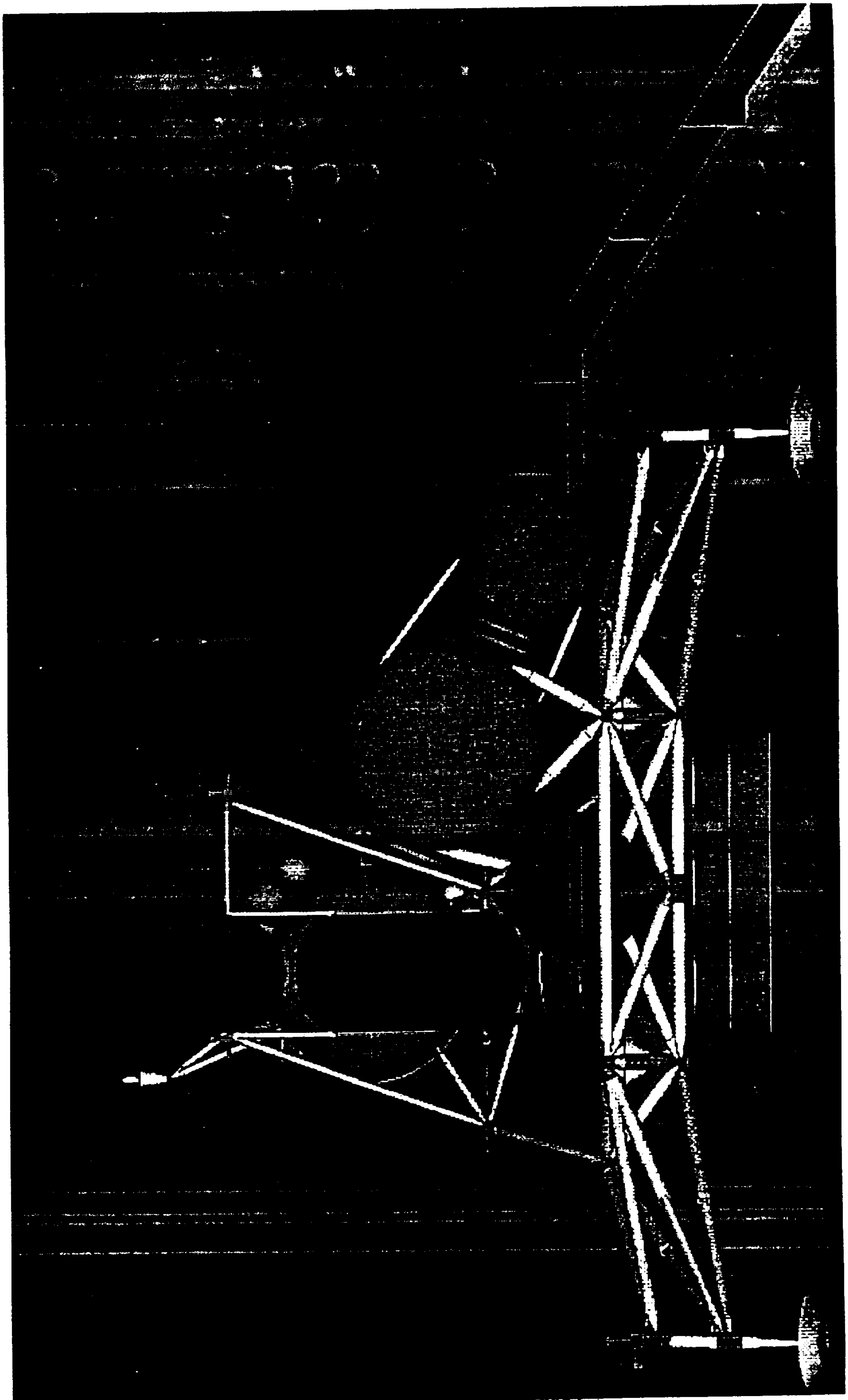
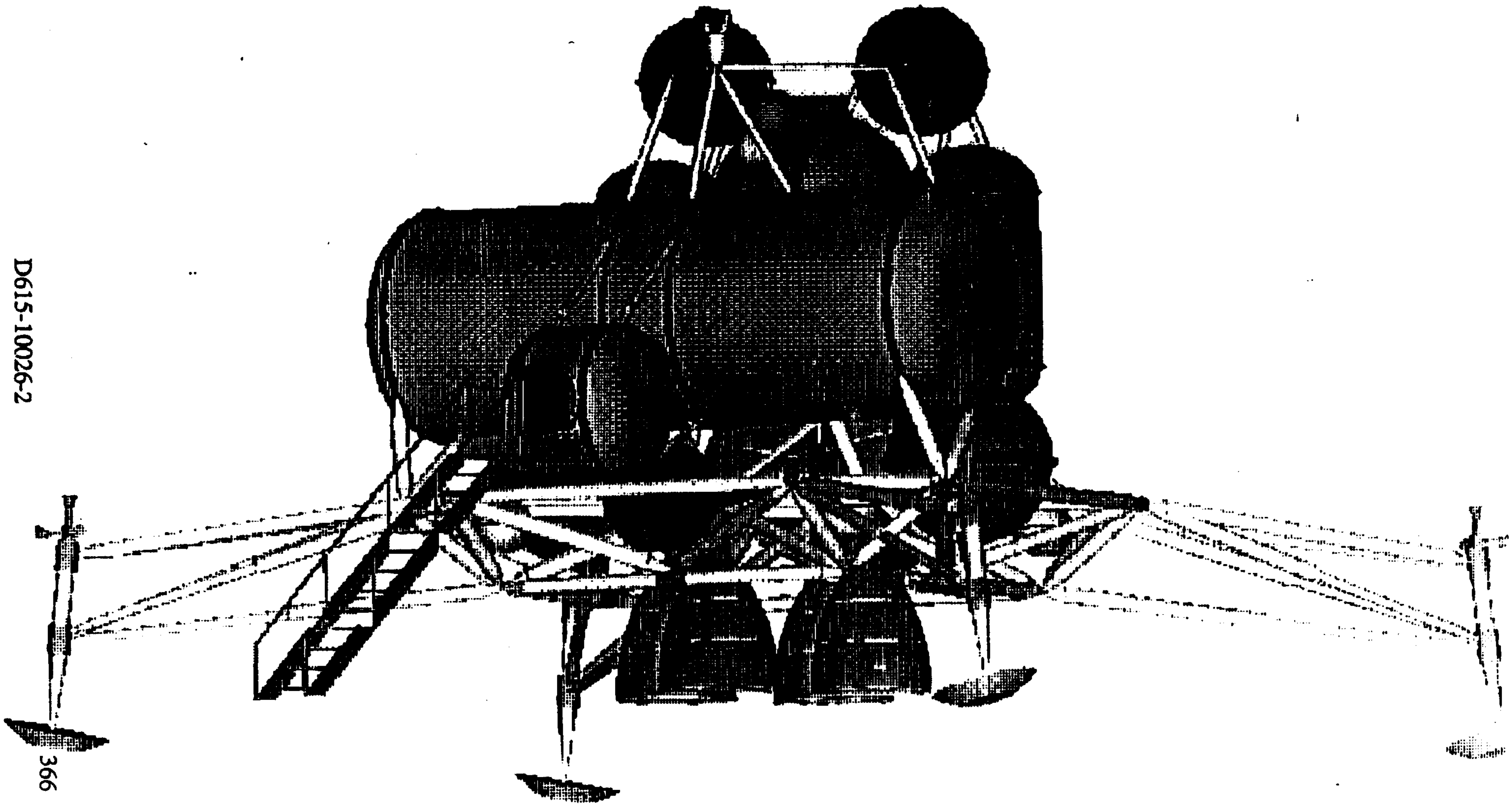
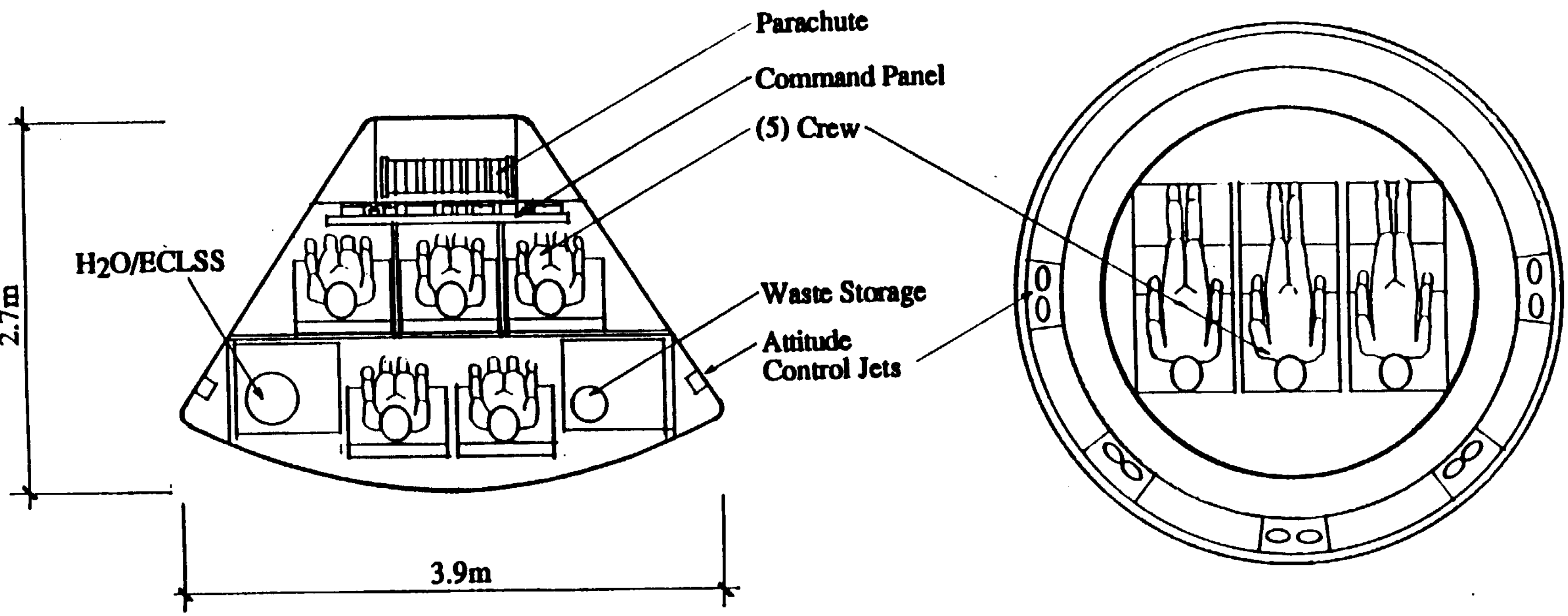


MEV Surface Configuration



Earth Crew Capture Vehicle (ECCV)



Cross Section

Top/Plan View

MTV Hab Trade Weight Groundrules

Primary Structure (trade discriminators)

- Pressure vessel
- Structure rings and ribs
- Pressure bulkheads (if any)

Secondary Structure (trade discriminators)

- Inter-module tunnels (if any)
- Inter-module integrating structure (if any)
- Pressure hatches separating redundant volumes
- Meteoroid, debris and thermal protection (surface-area-based)

Secondary Structure (not included; non-discriminators to first order)

- Airlocks
- Hatches associated with airlocks / EVA
- Windows
- Floors
- Walls
- Subsystem mounting standoffs

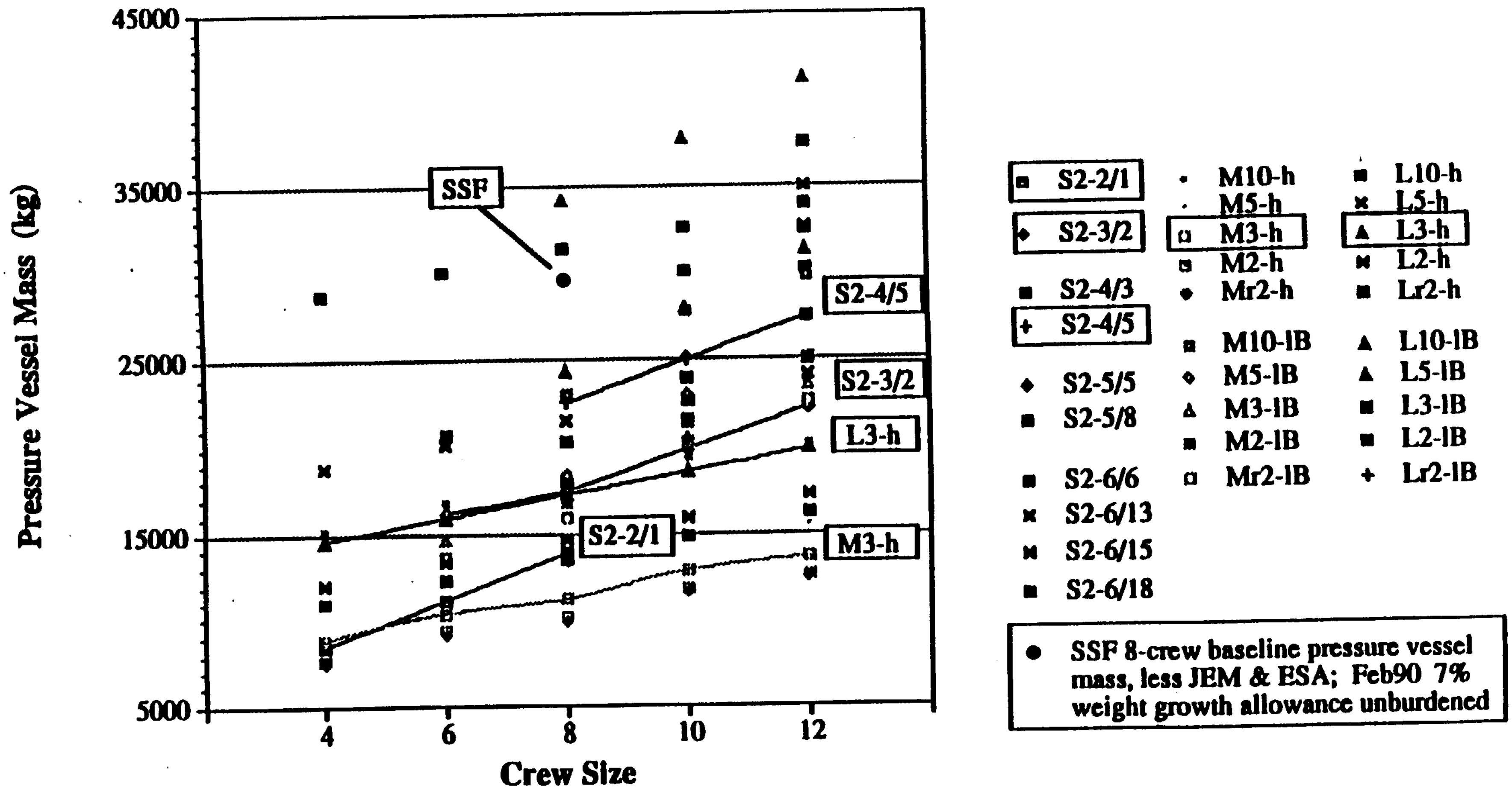
ADVANCED CIVIL SPACE SYSTEMS

Reference Concept Mass Analysis

BOEING
STCAEM/BS/24Feb90

Mass Sensitivity

Comparison of Reference Concepts



For crew sizes over 6, larger-diameter concepts have an increasing weight advantage over small-diameter cluster concepts

Module Concept Selection

The proportions of this module type do not approximate that of SSF modules until crew sizes of about 12 are reached. Beyond that point, it is useful to think of clustering these 7.6m-diameter modules together in simple topologies to extend the habitable domain, for surface bases as well as for large-crew in-space transportation systems.

Finally, it is important to remember that the nature of the trade study has led us to generate a quite conservative habitat concept, which although it combines features demonstrated to be advantageous, still reflects a rather limiting set of assumptions. As a next step, concepts should be considered which combine this reference module type with the smaller diameter module types which we still see as widely applicable throughout all phases of the HEI. For advanced applications, clusters which mix module types and sizes promise good accommodation of functional requirements as well as interesting and stimulating psychological environments.

Module Concept Selection

Selection

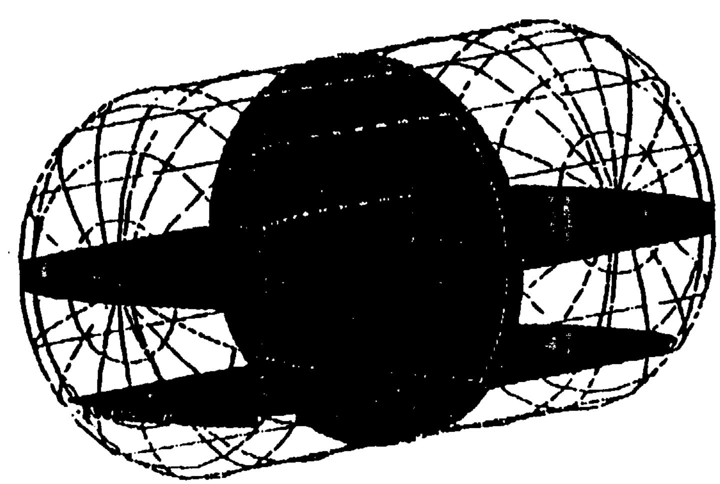
Modified Mg2-l concept family selected for further reference use in the STCAEM study, for

- concept development activities
- trade & sensitivity analyses
- more detailed habitat system definition

Major Features

- 7.6 m diameter
- 2:1 aspect ratio, unpenetrated end domes
- Cross - section, bisecting bulkhead
- Diametral tension - tie, deep floor
- Extensive commonality across architecture: g-field optimized

6 Crew Configuration



Module Concept Selection (2)

Functionality

- Unitary vessel minimizes leakage, parts count
- Permits wide variety of internal outfitting designs
- Diametral floor maximizes nominal floor area, facilitates weight-reducing tension tie
- Compact domain, good for access-time safety
- Best overall multi-floor efficiency in g-condition for a range of crew sizes
- Less wall area than smaller diameter; outfitting can compensate

Integration

- Minimizes orbital assembly operations required
- 7.6 m launch shroud likely available for early HEI
- Large crews can be accommdated through simple clustering
- Compact habitat facilitates aerobrake integration

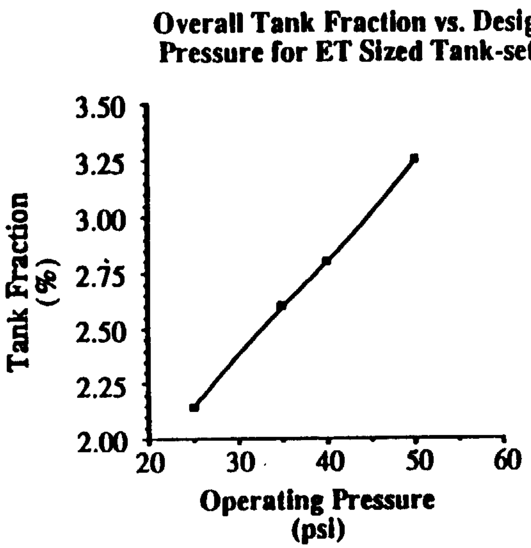
Perception

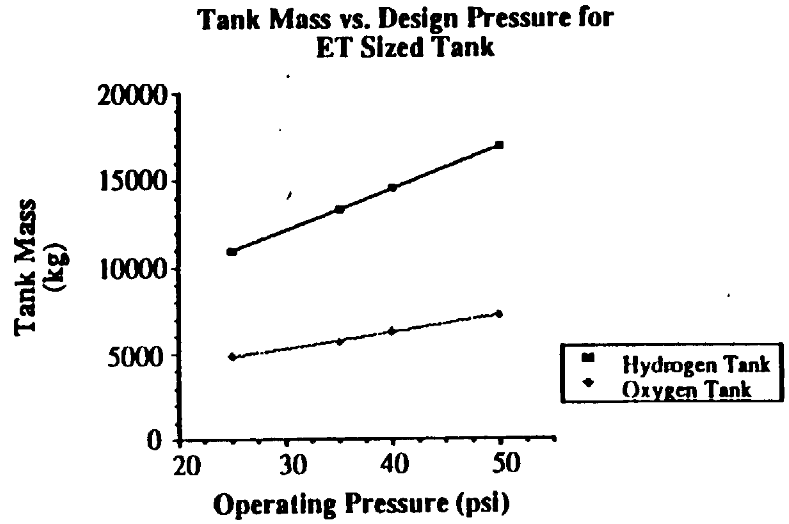
- Survey results show technical people perceive larger diameter concepts as more spacious
- Barrel vault proportionately invariant with crew (module) size, better than dome
- Module width has better plan aspect ratio than smaller diameters
- Low intrinsic number of unique spatial units; outfitting can compensate
- Lowest score for circulation option boredom over long duration

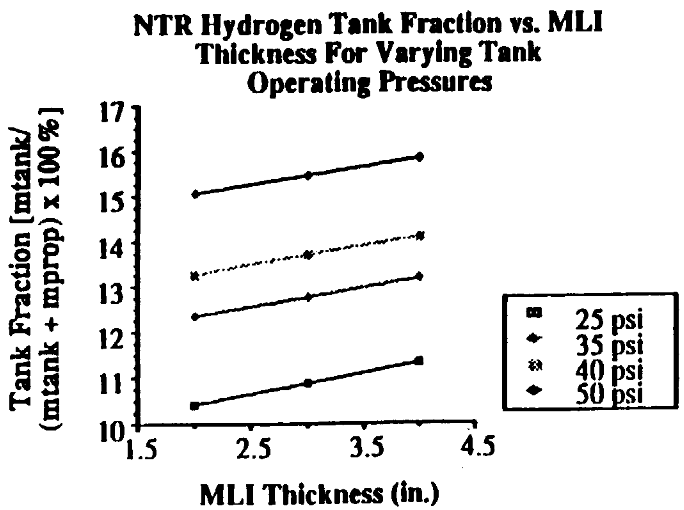
Cost

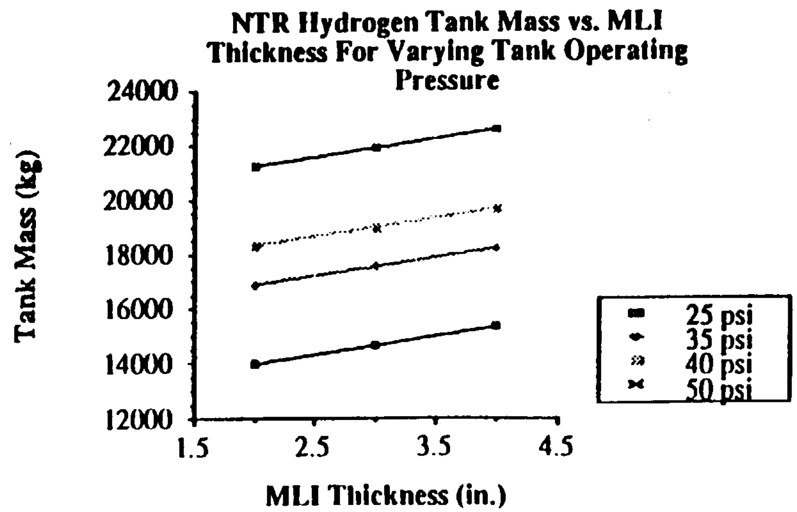
- Lightest weight (transportation cost critical for exploration vehicles)
- Welded-metal technology feasible here, well-understood
- Prime opportunity for M&P improvements, however
- End dome complication less than for 10 m size
- Commonality in growth architectures more appropriate for surface system applications

Cryo Boiloff Code Tank Properties Prediction for ET Sized Tanks









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Comparison of Predicted Tank Masses & Fractions With the External Tank

- External Tank Data -

Total Dry Mass = 35,425 kg
Propellant Load = 719,112 kg
  LH2 Load = 102,618 kg
  LOX Load = 616,493 kg
LH2 Tank Mass = 14,402 kg
LOX Tank Mass = 5695 kg
Tank Max. Operating Press:
  LH2 - 34 psi
  LOX - 22 psi

Overall Tankage Fraction = 4.7 %

LH2 Tank Fraction = 12.3 %
LOX Tank Fraction = 0.91 %

LH2/LOX Overall Tank Fraction = 2.72 %

- Boiloff Code Predictions -

Assumptions:
  MLI Thickness = 2"
  Diameter = 4.2 m
  Tank Shape - Cylindrical tank with $\sqrt{2}$ ellipsoidal endcones
  Ullage = 5 %
  Propellant Mass = ET Propellant Loads
  Mass includes vapor cooled shields, supports, and para-to-ortho H2 converter, where appl.

Results:

Tank	Pressure	Tank Mass	Tank Fraction (overall)	
H2	35 psi	13306 kg	11.5 %	(2.6 %)
O2		5730 kg	0.92 %	
H2	40 psi	14521 kg	12.4 %	(2.8 %)
O2		6229 kg	1.0 %	
H2*	50 psi	16951 kg	14.2 %	(2.99 %)
O2**	30 psi	5239 kg	0.84 %	

* NTR value-25 psi x 2 for overpress.
* 2 x overpress. allowance

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Major Propulsion Element List for 2000-2030 HEI program

Primary Objective: Furnish a top level list of all major propulsion elements necessary to a 3 decade HEI total program entailing Lunar, Mars opposition (short stay) and Mars conjunction (long stay) missions.

Secondary objective: Considering four candidate vehicle combinations (differentiated by propulsion system choice, each of which might satisfy all the space transfer objectives of a comprehensive HEI program) roughly evaluate or 'score' the *total development effort required to bring each propulsion system element/technology up to flight readiness*. Having done so, sum all the element scores for each of the candidate vehicle combinations in order to ascertain which combination meets HEI program objectives with least overall propulsion systems development effort. The 4 candidates are listed below:

(1) Chemical Lunar with chemical Mars opposition (zero-g) & conjunction (art-g,tether system)
(2) NTR Lunar, NTR Mars opposition (zero-g) & Mars conj (art-g, vehicle rotation about its Cg, no tether)
(3) Chemical Lunar, NEP Mars opposition (zero-g) & NEP conj (art-g, tether system)
(4) Chemical Lunar, SEP Mars opposition (zero-g) & SEP conj (art-g, tether system)

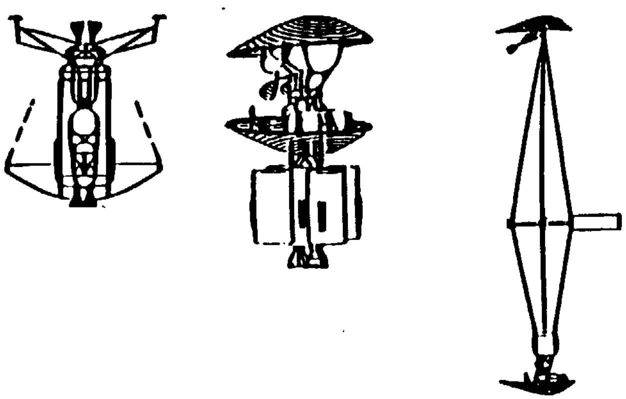
Scores: Primary list: the all NTR set had lowest total propulsion element count of 5, that is, 5 distinct elements were identified. The chemical/SEP combination followed with 8 elements, all chemical with 8, and chemical/NEP with 11. Differences in opinion as to what constitutes 'major' 'and 'distinct' propulsion elements might lead to slight variations in the totals, all depending on who does the counting.

Scores: Secondary list: The all NTR set scored the lowest in total propulsion elements development effort with a score of 13. the chemical/SEP combination and the all chemical set were about even with scores of 18 & 19 followed by chem/NEP at 27. These scores are relative, and only show how the 4 vehicle sets compare to one another; They are also subjective, and the differences in overall scores may be more pronounced, less pronounced or even change in rank depending on who is doing the evaluating. these rankings are not presented herein as the results of a precise technical trade study, but rather the results of a rough comparison 'methodology' with its major emphasis on a top down viewpoint in contrast to an analysis which as its emphasis on optimizing and/or selecting propulsion systems solely for individual missions.

Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

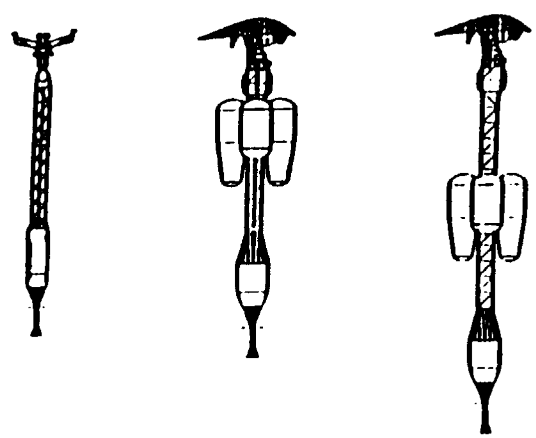
Moon	zero-g opposition (Mars)	artificial-g conjunction (Mars)	Propulsion Element	Development Effort Factor

Lunar Chemical / Mars Chemical sys



Propulsion Element	Development Effort Factor
Lunar	
1 LTV propul stg	2
2 LTV aerocapt brake	3
3 LEV propul stg	2
Mars zero-g vehicle	
4 MEV propul stg	2
5 MEV/MTV aerocapture brake	3
6 MTV propul stage	2
7 TMI propul stage	3
Mars artificial-g vehicle	
8 Art-g tether system	2
8 distinct propulsion elements with development factor scores:	**19**

Lunar NTR / Mars NTR sys



Propulsion Element	Development Effort Factor
Lunar	
1 LEV propul stage	2
Common Lunar & Mars zero-g	
2 Common LTV/MTV NTR propul stage	6
3 Radiation handling/monitoring/shield	2
4 MEV propulsion stage	2
5 MEV descent heat shield	1
Mars artificial-g	
no necessary additions	
5 distinct propulsion elements with development factors scores:	**13**

Legend: (1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to acheive flight readiness

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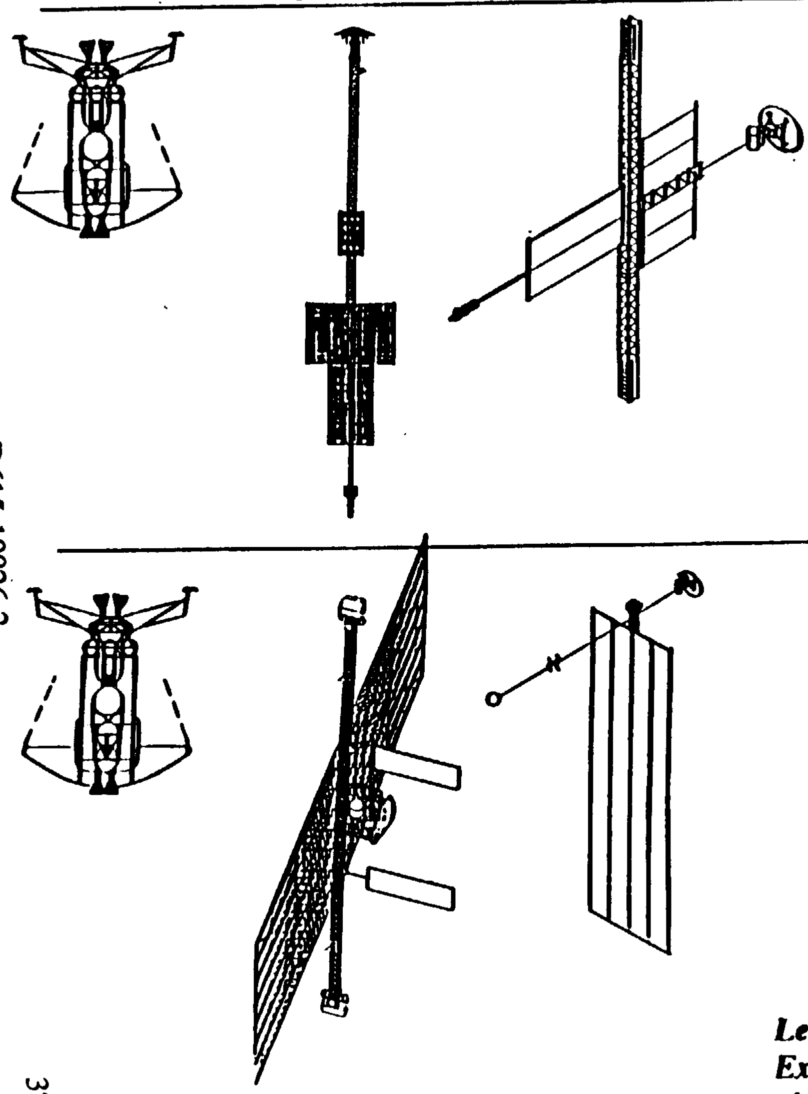
Major Propulsion Element List for Specific Vehicle Sets
to Satisfy Lunar & Mars Objectives of 2000-2030 HEI Program

ADVANCED CIVIL SPACE SYSTEMS — *BOEING*



Propulsion Element	*Development Effort Factor*
Lunar	*Lunar* *Chemical* / *Mars* *NEP*
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 NEP reactor	6
7 Radiation handling/monitoring/shield	2
8 Dynamic conversion equip	2
9 Radiatiors	2
10 Electric thrustors	3
- Separate crew carrier to NEP 'spirial up' altitude	0 (use LTV)
Mars Artificial-g vehicle	
11 Artificial-g tether system	2
11 elements w sum of devel factors scoring:	**27**
Lunar	*Lunar* *Chemical* / *Mars* *SEP*
1 LTV propulsion stg	2
2 LTV aerocapture brake	3
3 LEV propulsion stg	2
Mars zero-g vehicle	
4 MEV propulsion stg	2
5 MEV descent heat shield	1
6 SEP Solar array	3
7 Electric thrustors	3
- Separate crew carrier to SEP 'spirial up' altitude	0 (use LTV)
Mars Artifical-g vehicle	
8 Artificial-g tether system	2
8 elements w sum of devel factors scoring:	**18**

Legend:(1) least development effort; (6) most development effort
Expected total resources that must be expended for such a propulsion element to achieve flight readiness

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Weights Statement

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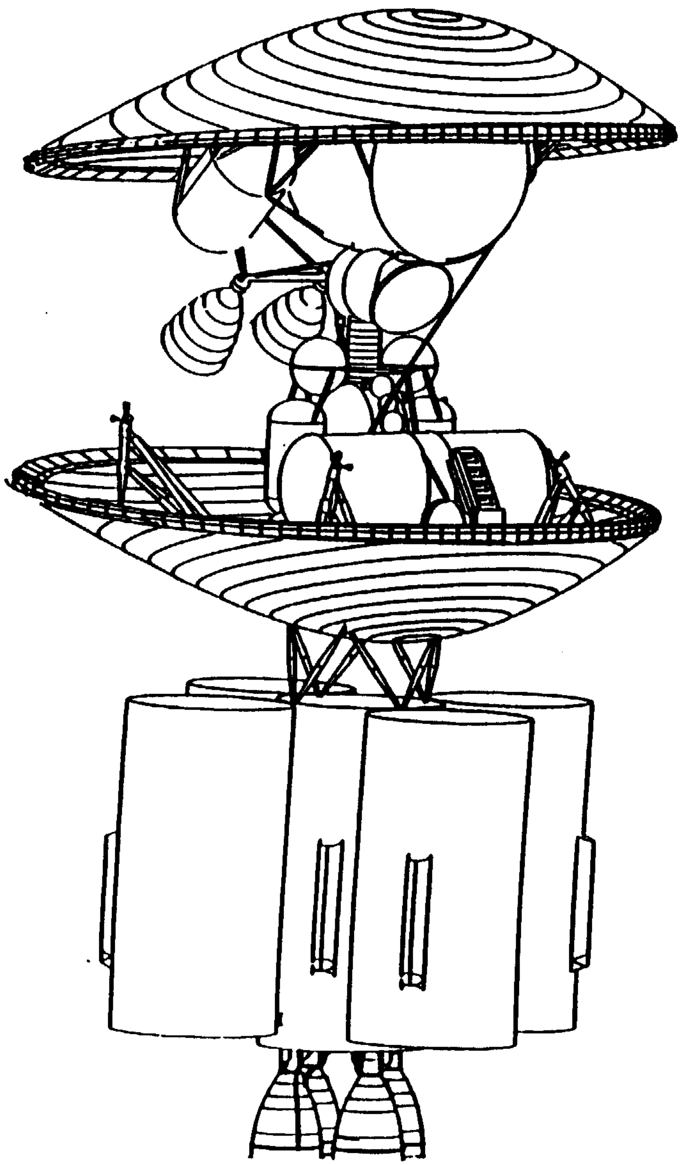
B. Weight Statements

Summary and detailed weight estimates are provided for the Chemical/aerobrake vehicle for the 2015 opposition mission opportunity. Assumptions made in the weight estimates include:

- Crew size of 4
- Use of Earth capture crew return vehicle
- Mission duration of 565 days.
- Improved technology (post-1990) for component weights (see technology section). The reference mass for this mission case is 800 tons in low Earth orbit.

Reference Chem/aerobrake Vehicle for 2015 Opposition Mission

Crew of 4, ECCV return, 565 day total trip time *Revision 2 5/22/90*



Element	*mass (kg)*
MTV Mars aerobrake	23758
MTV crew hab module 'dry'	28531
MTV consumables & resupply	7096
MTV science	1000
MTV propulsion stage	18206
MTV propellant load	85141
MTV total	163732
MEVMars capture & desc aerobrake	15138
MEV ascent stage	22754
MEV descent stage	21457
MEV surface cargo	25000
MEV total	84349
ECCV	7000
Cargo to Mars orbit only	0
MTV-TMI interstage wt	500
TMI inert stage wt	54560
TMI propellant load	490950
TMI stage total	545510
IMLEO	801090

Mac chart: M Ref chem/ab cover pg
synthesis model run# marschemmtv.dat;21

Ascent Cab - Ref MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 3 day occupancy time **Revision 2 5/22/90**

	Element	*mass (kg)*	*Rationale*
Cab ECLSS	Atmospheric Revitization Sys/ Trace contaminant control assembly	123	CO2 adsorption unit, expendable LIOH cartridge Pre & postsorbent beds, catalytic oxidizer for particulate & contaminant control
	Atmosphere Control System	62	Total & partial press control; valves, lines & resupply/ makeup O2 & N2 and tanks
	Atmos. Composition & Monitor Assem.	55	O2 & N2 monitor for ACS, particulate & contaminant monitor for ARS
	Thermal Control Sys	40	Temp control: sensible liq. heat exchanger, ext radiator wt included in 'secondary structure' mass
	Temp. & Humidity Control	240	Condensing heat exchanger, fans, ducting
	Water Recovery and Management	45	Stored Potable water only
	Fire Detection & Suppression Sys.	113	Automatic sys w manual extinquishers as backup
	Waste Management Sys and Storage	-	Considered part of 'Man Systems'
	Asc cab ECLSS mass	678	Apollo style open ECLSS system
Cab Structure	Primary/Secondary Structure	519	Overpressurized (20 psia) on launch for structural integrity. Stiffening rings added at cylinder/endcap interface for added strength. Skylab derived triangular grid floor with beam supports on 6" centers. Support ring interface on pressure vessel to carry loads imposed by the floor and equipment during launch to aerocapture.
	Berthing ring/mechanism (1)	139	
	Berthing interface plate (1)	90	
	Windows	90	
	Couches	80	
	Hatches (2)	80	
	Asc cab Structure mass	998	

synthesis model run number: marsntr.dat
Mac chart:M Ref MEVasc cab wt-ratio

/STCAPM/bhd/31May90

Asc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 30 day stay, 2 advanced space engines; Isp=475 sec *Revision 2 5/22/90*

Group	Ref	Item	Mass	Notes
Ascent Cab		Structure	998	SSF dia center cyl section w ellip ends. Stiffening rings added. See 'Structures pg'
		ECLSS	678	Open sys:CO2 adsorption unit, stored H2O,O2,N2, no airl., no hyg w. see 'ECLSS pg'
		Command/Control/Power	330	Power: fuel cells
		Man systems	82	Waste management sys/waste storage/medical equp.
		Spares & tools	192	Subsystem component level spares
		Wt growth	376	15% growth for dry mass
		Asc 'dry' mass	2656	Total cab dry mass
		Consumables (food & water)	62	Minimum; food and water only; 3 occupancy
		Crew/effects/EVA suits	760	Crew of 4, 100 kg EVA suit per crew member
	[22]	*Ascent cab gross mass*	3478	
			Fuel / Oxidizer	
Ascent stage inert	[45/46]	Single tank wt	312/140	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[68/69]	Meteoriod Shield	40/18	One 0.40 mm sheet of Al
	[50/51]	MLI	59/26	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	71/72/112/113	VCS & Vacuum shell	47/21	1 VCS and 1 Vac shell: both 2 x 0.13mm Al outer sheet w 0.57kg/m2 honeycomb core
	[2x1316]	Propel line wt	50/50	50 kg per tank
	[116/117]	Tank wt growth	62/31	15% wt growth
	[73/74]	Sum single tank inerts	617/307	Total tank & tank inert wt
	[114/115]	*Tot: H2 & O2 tanks:*	1234/614	2 LH2 & 2 LO2 tanks
	[500]	Main propulsion	564	3 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[118]	Asc frame & struc wt	478	3% of total asc stg propellant wt
	[1274,525]	RCS inert	122	Estimate from RCS prop load
	[54]	Propul, frame wt growth	174	15% of total inerts
	Sum	*Asc propul & frame inert*	1338	
Prop loads	[60]	Asc usable propellant	15500	*Asc veh dV= 5319 (m/sec) to 250 km periapsis alt. by 1 sol orbit.*
	[56+58]	Asc boiloff	418	50 day sep from MTV before M arr+ 30 day surf stay;calc:Boeing 'CRYSTORE' program
	[52]	Asc RCS prop	172	N2O4/MMH prop, Isp=280 sec, *Asc RCS dV =35 (m/sec)*
	Sum	*Total Asc propellant load*	16090	
	[63]	*Asc veh total mass*	22754	all masses in kg

synthesis model run#: marslander.dat;108

Mac chart: M Ref MEV asc veh wt-rationale STCAEM/bbd/22May90

Desc stage - Reference MEV for 2015 Chem/Aerobrake Vehicle

Crew of 4, 30 day stay, 4 advanced space engines; Isp=475 sec, 25 t surf cargo

Revision 2 5/22/90

Group	Ref	Item	Fuel / Oxidizer	Notes
Desc stage inert	[98/99]	Single tank wt	242/126	2 SiC/Al metal matrix tanks for each, 37ksi wk stress, MEOP=175 kPa, min t=3.5mm
	[124/125]	Meteoriod Shield	31/16	One 0.40 mm sheet of Al
	[122/123]	MLI	47/24	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm.
	[126/127]	Vapor Cooled Shields	37/19	1 VCS at 2 x 0.13mm Al outer sheet w 0.57 kg/m2 honeycomb core
	[0/0]	Vacuum shell	0/0	not on desc tanks
	[2x1316]	Propel line wt	50/50	50 kg per tank
	[132/133]	Tank wt growth	41/23	15% wt growth
	[128/129]	Sum single tank inerts	448/258	Total single tank + tank inert wt
	[130/131]	***Tot: Fuel & Ox tanks:***	896/516	2 LH2 & 2 LO2 tanks
	[501]	Main propulsion	1127	4 x 30klbf Adv eng's: Isp=475 sec, w extendible/retractable nozzles
	[102]	Asc frame & struc wt	567	4% of desc stage stg wt + 2% of surf crew mod mass
	[103]	Landing legs	1487	3% of total landed mass
	[1273,526]	RCS inert	331	Estimate from RCS prop load
	[104]	Propul, frame wt growth	490	15% of total inerts
	Sum	***Desc propul & frame inert***	4002	
Prop loads	[91+92]	Desc usable Prop	13477	*Desc propulsive veh dV= 931 (m/sec) from 250 km periapsis alt. by 1 sol orbit.*
	[0]	Desc boiloff	0	
	[101]	Desc RCS prop	2566	
	Sum	***Total Desc propellant load***	16043	N2O4/MMH prop, Isp=280 sec, *desc RCS dV=100 (m/sec)*
Aero brake wt		***MEV aerobrake:***		Structural design assumptions:
		• Primary spar wt	2484	200ksi spar strength
		• Secondary spar wt	2596	22.5 inch spar depth
		• Honeycomb wt	6758	note: 200ksi may require additional material technology developement efforts
		• TPS wt	3300	
	[78]	Total:	15138	
	[77]	***Surface crew hab module***	25000	Level II Requirement: surf modulw, surf science & surf stay consumables
	[61]	***Asc veh total mass***	22754	from 'Asc stage' wt statement page
	[106]	***MEV mass***	84349	all masses in kg

synthesis model run#: marslander.dat.1

STCAEM/bbd/23May90 Mac chart: M Ref MEV decs veh wt-ration

Crew hab mod - Reference MTV for 2015 Chem/aerobrake Vehicle

Crew of 4, 565 day total trip time *Revision 2 5/22/90*

	Element	*mass (kg)*	*Rationale*
[360]	Structure	8351	Cyl length: 9 m, dia: 7.6 m, ellip ends, 3 levels; tri grid w beam supports. Tens. ties
[363]	ECLSS	4256	SSF derived with same degree of closure, sized for crew of 4 for 565 days
	Command/Control/Power		
[364]	• Internal	1159	ECWS, DMS, batteries, other avionics/computing/monitoring eq, conditioning equip
[1281]	• External Power	1539	Solar array,boom, power distribution,power management,fuel cell system
[368-316]	Man systems	4121	Wts -all sys:SSFderived(as a funct. of crew size&occupancy time)for Mars missions
[316]	Crew & effects	440	110 kg per person including personal belongings
[373]	Spares/Tools	1496	Subsys component level spares. Life crit sys are 2 fault tolerent (approach of SSF)
[247]	Radiation shelter	1802	Provides 10 g/cm2 protection + 3-5 g/cm2 provided by vehicle structure and equip
[377]	Weight growth	2973	15% weight growth for dry mass excluding crew & effects and radiation shelter
[378]	Airlocks	1530	2 x 765 kg external airlocks (shuttle type airlocks modified for MTV mission)
[330]	EVA suits	0	EVA suits weight counted in MEV ascent cab weight statement
[160]	TTNC & GN&C platforms wt	863	3 platforms
Sum	***MTV 'dry' crew hab mod wt***	28531	'dry' hab module represents structure and support systems equip & hardware that are dependant on crew size and independant of mission duration
[371]	*On board equip resupply	1304	Based on adjusted SSF resupply reqts for pot w, hyg w, ARS,TCS/THC &WMS
[398]	*Consumables	5792	Crew of 4 for 565 days; food:2.04 kg/man/day, food pkg:0.227, pharmaceuticals: 0.25 other: 0.291 Clothes: 42 kg/man. food vol: 0.0055 m3/man/day, other: 0.0018.
[380]	***MTV crew mod 'wet' wt***	7096	
[165]	*Transfer science equipment	1000	Inb and outb MTV science hardware and supplies
	Remote Manipulator-arm Sys	0	all large external self assembly hardware left in LEO
380+179	***MTV crew mod & support systems weight***	36627	This wt refects the Boeing ref crew of 4 mod loaded for the 2015 opposition mission. The mod 'dry' wt represents a SSF type closed ECLS Sys (air >99%, water >95%) that serves the crew with 2 fault tolerence on all life critical sys except structure. Its wt varies primarly with crew size. consumables wt varies with crew size and mission duration.

** MTV hab mod consumables, resupply, and transit science dependant on mission duration, and free abort requirement. I.e. crew mod 'wet' wt will vary for different missions*

Mac chart: M Ref MTV mod wt-rationale
synthesis model run# marschemmtv.dat;21

Mars departure stg - Reference MTV for 2015 Chem/Aerob Veh

Crew of 4, 2 advanced space engines; Isp = 475 sec

Revision 2 5/22/90

Group	Ref	Element	mass (kg)	Rationale
TEI stage inert	[154]	Fuel tank	5424	2 SIC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[155]	Oxygen tank	3100	2 SIC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[158]	MLI/meteor shield	1082	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor Shield: 2 (kg/m2)
	[159]	Frame structure	5132	5% of MTV propellant + 5% of MTV stg inert mass
	[518]	Main propulsion	794	2 x 30k lbf advanced space eng's: Isp=475 s, high AR nozzle not extendible
	[183]	RCS inert	300	Scaled from RCS propellant
	[160]	Mass growth	2374	15% growth for inert stage
	[161]	*Mars dep stg 'dry' wt*	18206	
TEI prop, boil-off & totals	[118]	MTV RCS propellant	699	Storable: N2O4/MMH propellant, Isp=280 sec, MTV RCS dV=30 m/sec
	[122]	MTV inb midcourse burn prop	1256	delta V: 90 (m/sec); burn done with MTV Mars dep main propulsion
	[128]	Mars dep usable prop	71525	LH2/LO2, MR=6:1, *Mars dep dV: 3400 m/sec* usable=prop req after outb & inorbit
	[545+546]	In orbit Mars dep prop boiloff	426	boiloff; 30 day boiloff period; calculated with Boeing's 'CRYSTORE' program
	[554]	*tot onboard prop at Mars arr*	73906	
	[121]	Outb midcourse burn prop	6709	midcourse maneuver delta V: 120 (m/sec); burn done w MTV main propulsion
	[498+499]	Outb Mars dep prop boiloff	4526	335 day outbound trip time.
	[555]	*MTV propel expended outb*	11235	
	[556]	***Tot M dep propulsive stg wt (at time of E dep burn)***	103347	

synthesis model run #: marschemmtv.dat;21

Mac chart M Ref MTV vch wt-rationale

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TMI stg - Reference MTV for 2015 Chem/Aerobrake Veh

ECCV Return, 4 x 200k lbf advanced space engines; Isp = 475 sec

Revision 2 5/22/90

		Element	*mass (kg)*	*Rationale*
TMI Pay load	[556]	Tot MTV Mars dep stg	103347	See mars dep stage wt statement
	[380+179]	MTV Crew hab mod sys	36627	See MTV crew hab module wt statement
	[230]	ECCV	7000	4 man apollo type entry vehicle; MTV expended
	[106]	MEV	84349	4 man, 30 day stay, 25 t surface cargo
	[159]	Outb 'to-Mars-orbit' cargo	0	communication sat's taken on precurser mission
	[1292]	Mars Site Recon Vehicle	0	Not taken for Ref 2015 mission
	[163]	MTV-TMI interstage wt	500	Structural member joining TMI to MTV
		MTV Mars capture aerobrake:		Structural design assumptions:
		• Primary spar weight	4239	200ksi spar strength
		• Secondary spar wt	3434	22.5 inch spar depth
		• Honeycomb wt	12785	note: 200ksi may require additional material technology development efforts
		• TPS wt	3300	
	[169]	Total:	23758	
	[168]	***Tot TMI stg 'Payload wt'***	255581	TMI propulsive stg injects this wt into hyperbolic trajectory
TMI stage	[172-173]	TMI stage inert	54560	0.9 propellant fraction
	[173]	TMI propellant load	490950	TMI stage tanks topped off before ignition, no boiloff accounted for
	[172]	***TMI stage total mass***	545510	4 x 200k lbf advanced spaace engines, Isp=475 sec
	[171]	***IMLEO*** *Initial mass in low Earth orbit*	801090	

synthesis model run #: marschemmtv.dat;
Mac chart M Ref TMI wt-rationale

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Artificial Gravity Option

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Cryogenic/Aerobrake Vehicle
Artificial Gravity Configuration

The cryo/ab artificial gravity configuration employs a tether to achieve the radius desired to spin the transfer habitat at 56 m and 4 rpm to produce 1g. The tethers used are conductive tethers to avoid having separate power lines running in conjuction with the tether, thus complicating the reeling cycles. The conductive tether used is "ribbon" shaped to avoid entanglement during the reeling cycles, to better facilitate "crawler" operations, and because it radiates conductive heat better due to increased surface area over a circular cross-section.

The configuration is a 3 tether planar beam configuration with the crawler, solar arrays and communications laser located at the CM. The vehicle separates post-TMI with the transfer hab and MEV contiguously connected and the MTV aerobrake and TEI propellant used as countermass. The Mars to Earth configuration uses the MTV aerobrake and the empty TEI propellant tanks as countermass which results in a longer countermass radius to keep the transfer habitat at 56 m. If the MTV aerobrake is jettisoned at Mars in a nonreusable scenario, the Mars to Earth countermass radius would increase substantially to over 2 km.

The crawler/mast/power configuration at the CM of the vehicle is deployed on trusses that package into the crawler assembly. The solar array and the communications laser are on despun joints for tracking, and the entire assembly packages below the transfer habitat in the MTV. The crawler is divided into 2 sections so that one section can always be at the CM to support the deployable truss and the tether. The crawler taps into the aluminum conductor to transfer power from the solar array to the crew systems. Each crawler section has 2 small solar arrays for independent power during movement along the tether.

The cryo/ab mass penalty, when compared to a reusable 0g version, is ~ 15%, because of the hardware and propellant required to support artificial gravity operations. The MTV aerobrake would have to increase in size from 30 m to 32 m to accommodate packaging of the tether reel, crawler, solar arrays, and communications laser below the transfer habitat. 2 despun joints are also required for the solar array and communications laser.

Artificial Gravity (g_a) Assessment Assumptions

A 1g gravity level was assumed for this study over partial g because the minimum gravity level required to offset physiological deterioration is not known. The rotation rate was set to be no more than 4 rpm, which is based on experimental data in the Pensacola Slow Rotation Room (1960's) on human adaptation. The crew compartments are contiguously pressurized during all mission phases, and the crew modules are to be oriented with the long axis parallel to the spin vector to offset the Coriolis effect along major circulation paths. Connections between habitation and the countermass are either tethers or a truss rather than a pressurized tunnel because, since all crew compartments are contiguous, the is no need for an IVA transfer.

Artificial Gravity (g_a) Assessment

Assumptions	Rationale
1g gravity level	• Earth-normal conditioning for exploration in surface EMU
Rotation rate ≤ 4 rpm (56 m)	• Generally accepted range for vestibular disturbance tolerance
Contiguous crew compartments	• Maximize available volume • In-flight simulation and training • Contingency operations
Truss and tether connections • Tethers are "ribbon" shaped	• Avoids mass penalty • Not needed for contiguous volumes • Facilitates conductors
Module orientation parallel to spin vector	• g level consistency; minimizing vestibular disturbance • Mass properties quasi-isotropic to first order

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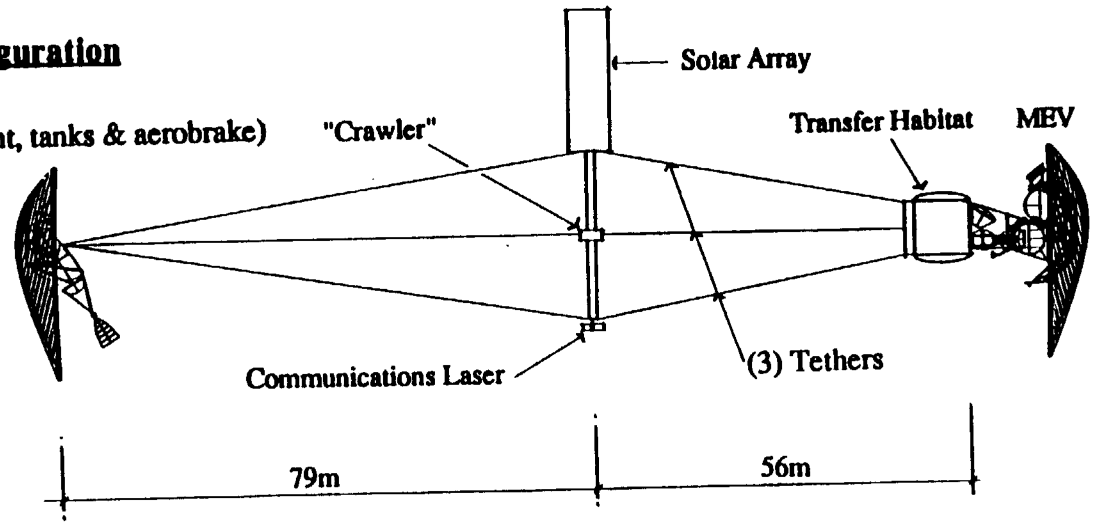
g_a Cryo/AB Configuration

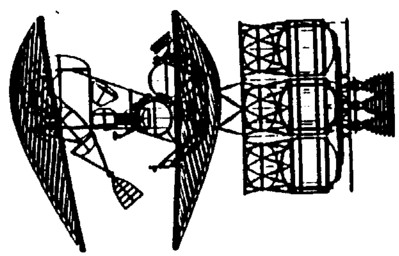
This chart shows the 3 main configurations of the vehicle in transit. The Earth to Mars phase requires a total tether length of 128m, while the Mars to Earth phase requires a total tether length of 161m with MTV aerobrake, and 2.15km without MTV aerobrake. The solar array and the communications laser are located at the CM on a "despun" joint to track the sun and Earth respectively. The crawler is also located at the CM nominally, but has the ability to travel to either end of the central tether to transfer crew and/or supplies. The initial TMI configuration is shown for comparison.

g_a Cryo/AB Configuration

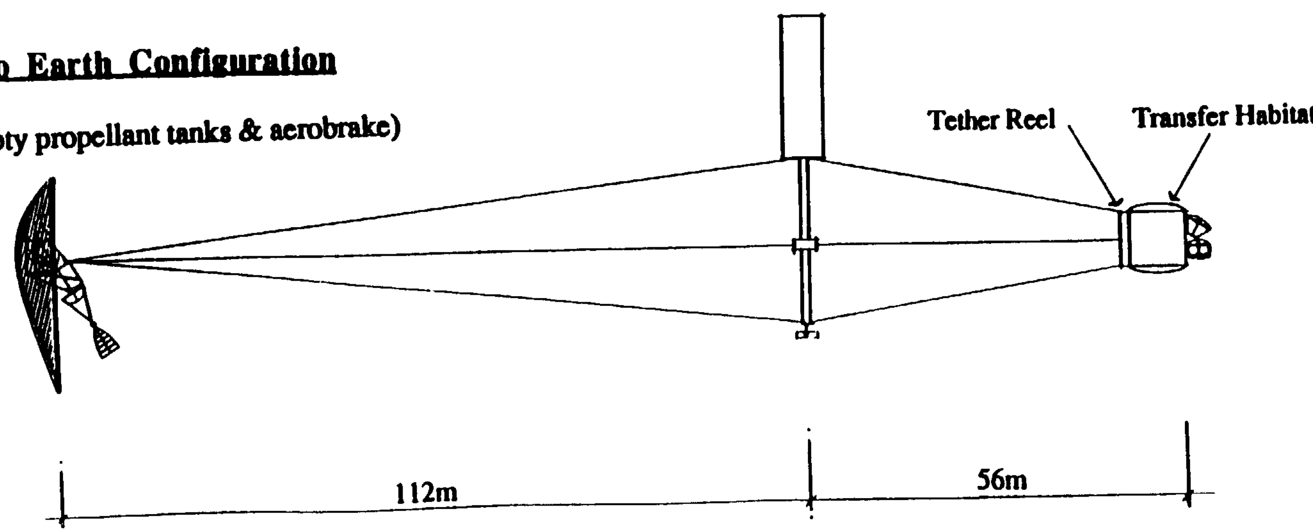
Earth to Mars Configuration



TMI Configuration



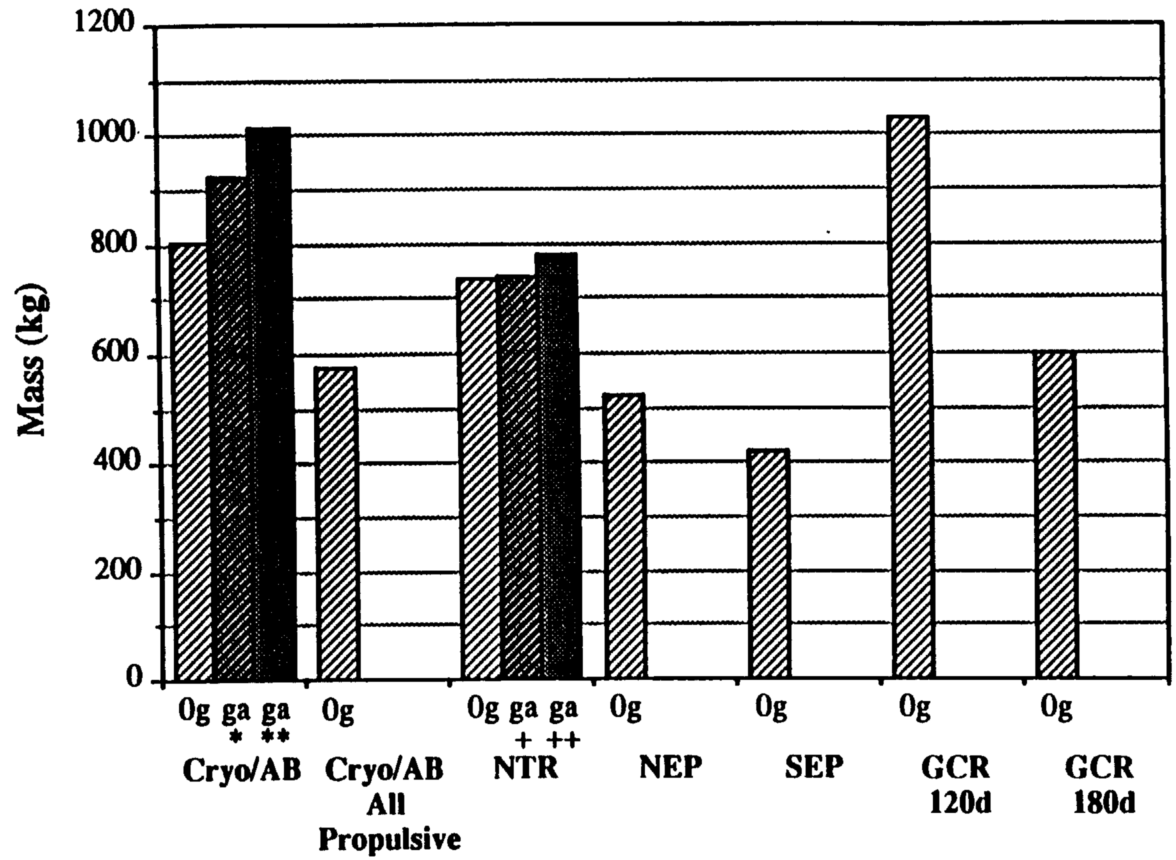
Mars to Earth Configuration



g_a Mass Summary

This chart shows the relative mass of the Cryo/AB and the NTR artificial gravity configurations as compared to all the reference 0g configurations. The Cryo/AB configuration trades very poorly in artificial gravity, whereas the NTR configuration has only minor mass impact.

g_a Mass Summary



* ECCV crew return, MTV aerobrake not used as return countermass
** Earth aerocapture, MTV aerobrake used as return countermass
→ Level II 2015 565d option

\+ 1/3g NTR option
\++ 1g option
→ Boeing nominal 2016 434d option

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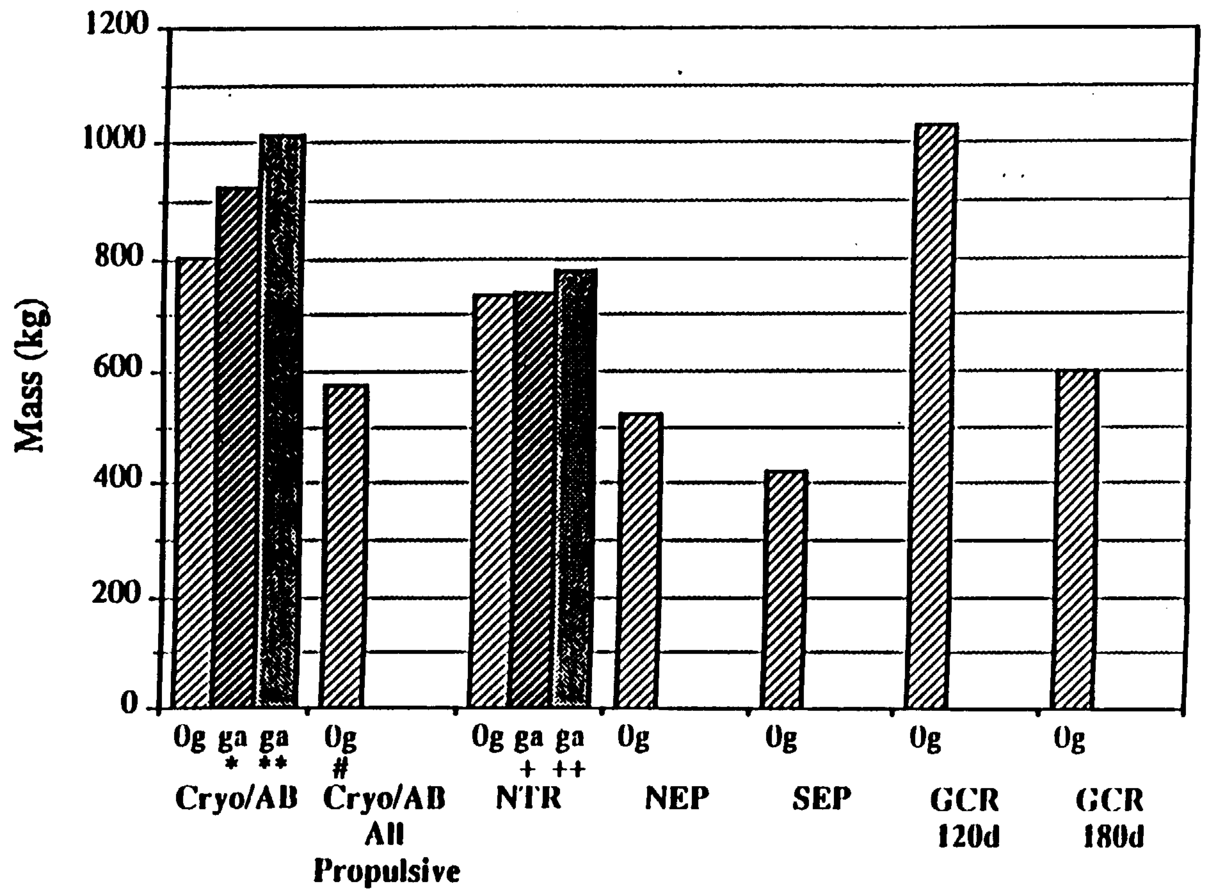
Artificial Gravity (g_a) Assessment Assumptions

- Gravity level
  - 1g chosen over partial g (less than 1g)
- Rotation rate
  - $\leq$ 4 rpm (4 rpm at 56 m nominally)
- Crew compartments
  - contiguously pressurized throughout all mission phases
- Connection
  - truss and tethers rather than a pressurized tunnel
  - multiple tethers are used that are "ribbon" shaped in cross section
- Module orientation
  - long axis parallel to spin vector

g_a Mass Summary



* ECCV crew return, MTV aerobrake not used as return countermass
** Earth aerocapture, MTV aerobrake used as return countermass
Level II 2015 565d option

\# Represents a conjuntion class mission, thus the reduced mass

\+ 1/3g NTR option
++ 1g option
Boeing nominal 2016 434d option

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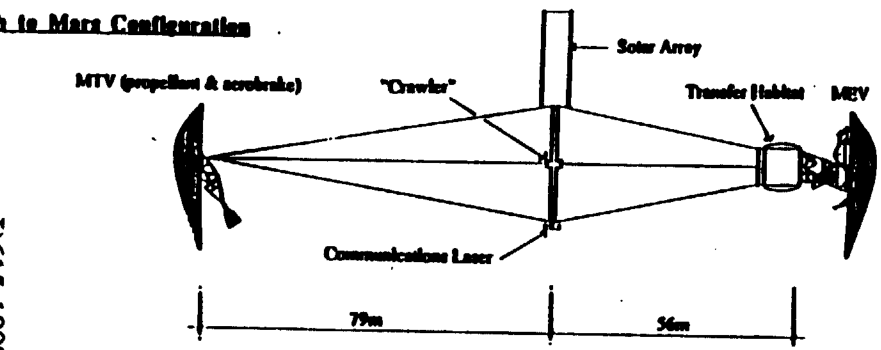
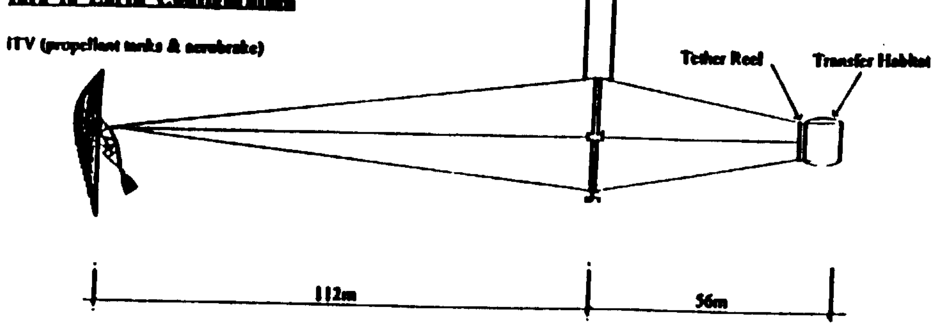
g_a Cryo/AB Mass Statement

Earth to Mars Configuration



Mars to Earth Configuration



Element	*ECCV return*	*Earth Aerocapture*
MTV Mars aerobrake	23758	23758
MTV crew hab module 'dry'	28531	28531
MTV consumables & resupply	7096	7096
MTV science	1000	1000
MTV propulsion stage	21847	27473
MTV MPS propellant load	103129	132177
*MTV Art-g added hardware wt	8057	8057
*MTV Art-g added RCS prop	9004	10627
MTV total	202422	238719
MEV Mars capture & desc aerobrake	15138	15138
MEV ascent stage	22754	22754
MEV descent stage	21457	21457
MEV surface cargo	25000	25000
MEV total	84349	84349
ECCV	7000	0
Cargo to Mars orbit only	0	0
MTV-TMI interstage wt	500	500
TMI inert stage wt	62820	69080
TMI propellant load	565270	621540
TMI stage total	628090	690620
IMLEO	922,361	1,014,188

** Artifical -g system weight penalties* — *all masses in kg*
Earth aerocapture configuration shown in diagram

Mac chart: M 1-g chem/ab cover pg
synthesis modelrun#marschemmtv.dat;33&34

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TMI stg - MTV for Artificial-g (1-g) 2015 Chem/Aerobrake Veh

4 RPM, 4 spinup/down maneuvers, ECCV Return, 4 x 200k lbf adv eng's: Isp = 475

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		Element	*mass (kg)*	*Rationale*
TMI Pay load	[556]	Tot MTV Mars dep stg	142037	See mars dep stage wt statement
	[380+179]	MTV Crew hab mod sys	36627	See MTV crew hab module wt statement
	[230]	ECCV	7000	4 man apollo type entry vehicle; MTV expended
	[106]	MEV	84349	4 man, 30 day stay, 25 t surface cargo
	[159]	Outb 'to-Mars-orbit' cargo	0	communication sat's taken on precurser mission
	[1292]	Mars Site Recon Vehicle	0	Not taken for Ref 2015 mission
	[163]	MTV-TMI interstage wt	500	Structural member joining TMI to MTV
		MTV Mars capture aerobrake:		Structural design assumptions: 200ksi spar strength, 22.5 inch spar depth
		• Primary spar weight	4239	
		• Secondary spar wt	3434	
		• Honeycomb wt	12785	
		• TPS wt	3300	
	[169]	Total:	23758	
	[168]	***Tot TMI stg 'Payload wt'***	294271	TMI stage injects this 'payload' wt into Mars hyperbolic trajectory
TMI stage	[172-173]	TMI stage inert	62820	0.9 propellant fraction
	[173]	TMI propellant load	565270	TMI stage tanks topped off before ignition, no boiloff accounted for
	[172]	***TMI stage total mass***	628090	4 x 200k lbf advanced spaace engines, Isp=475 sec
	[171]	***IMLEO*** ***initial mass in low Earth orbit***	922361	

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Mac chart M 1-g TMI wt-rationale

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Mars dep stg - for Artificial-g (1-g) 2015 Chem/Aerob Veh

4 RPM tether system, 4 spiuup/down maneuvers, Crew of 4, 2 adv eng's; Isp = 475

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Group	Ref	*Element*	*mass (kg)*	*Rationale*
TEI stage inert	[154]	Fuel tank	6545	2 SiC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[155]	Oxygen tank	3769	2 SiC/Al metal matrix tanks, 37 ksi working stress, tank MEOP=175kPa
	[158]	MLI/meteor shield	1282	MLI: density = 32 (kg/m3); 100 layers at 20 layers/cm. Meteor Shield: 2 (kg/m2)
	[159]	Frame structure	6243	5% of MTV propellant + 5% of MTV stg inert mass
	[518]	Main propulsion	794	2 x 30k lbf advanced space eng's: Isp=475 s, high AR nozzle not extendible
	[183]	Maneuver RCS inert	300	Scaled from RCS propellant
	[160]	Mass growth	2914	15% growth for inert stage
	[161]	***Mars dep stg 'dry' wt***	21847	
Art-g spin equip		Added wt necessitatedby Art-g		
		• tethers	2178	3 at 180 m each
		• tether reel	3050	Attached to MTV hab
		• tether crawler	1500	Transverses tether length, centers solar arr at veh/tether sys Cg for despun operation
		• lock joints & other equip	829	Lock joint secures MTV mod to MTV propulsion stg during MPS thrusting
		• added Spin RCS sys	500	Spinup/down RCS thrusters, lines, tanks, etc above the nominal maneuver RCS
		Total Art-g only hardware	8057	Total Art-g hardware penalty not required for a zero-g vehicle
Art-g spin prop		Art-g spin RCS propellant		1-g Art-g; See diagram of inflight spinup phases.
	[224/225]	• outbound 1st spinup/down	3298	Each of 2 counter wts (hab mod+MEV & MTV 'wet' propul stg+AB) spun to 4 RPM
	[222/223]	• outb 2nd spinup/down	3258	Despun for outbound midcourse corection MPS burn, then respun to 4 RPM
	[220/221]	• inb 1st spinup/down	1224	MEV left behind, MTV propul stg 'dry' except for inb midc correction burn
	[220/221]	• inb 2nd spinup/down	1224	
	Sum	**Total Art-g only RCS prop**	9004	Total Art-g RCS propellant penalty
TEI prop & boil-off	[118]	RCS maneuver propellant	599	Gaseous O2/H2 propellant, Isp=400 sec, MTV maneuver RCS dV=30 m/sec
	[122]	MTV inb midcourse burn prop	1516	delta V: 90 (m/sec); burn done with MTV Mars dep main propulsion
	[128]	Mars dep usable prop	87732	LH2/LO2, MR=6:1, *Mars dep dV: 3400 m/sec* usable=prop req after outb & inorbit
	[545+546]	In orbit Mars dep prop boiloff	485	boiloff; 30 day boiloff period; calculated with Boeing's 'CRYSTORE' program
	[121]	Outb midcourse burn prop	7638	midcourse maneuver delta V: 120 (m/sec); burn done w MTV main propulsion
	[498+499]	Outb Mars dep prop boiloff	5159	335 day outbound trip time.
	Sum	***tot MTV propellant load***	103129	
	[556]	***Tot M dep propulsive stg wt (at time of E dep burn)***	142037	

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Mac chart M 1-g MTV veh wt-rationale

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g_a Cryo/AB Vehicle Features

- Nominal spin rate = 4 rpm (56m to create 1g)
- Conductive tether
- Sun-tracking solar arrays
- "Crawler" contingency for crew transfer from end to end
- Nominally 4 spin-up/spin-down cycles (1 for conjunction class mission)
  - Outbound - MTV aerobrake and propulsion as countermass
  - Inbound - Empty MTV propulsion and aerobrake as countermass (MEV expended)
    - MTV aerobrake not required because of ECCV; however, it is useful as a countermass and will be retained for fully-reusable mission modes

g_a Cryo/AB Penalty Assessment

- Added mass
  - (3) tethers
  - Tether reel
  - Tether crawler
  - Added solar array
  - Added communications laser
  - Lock joints for transfer hab
  - Added RCS and propellant
  - Added TMI/TEI propellant

- MTV aerobrake
  - 2 m larger than MEV aerobrake
    - needed due to packaging constraints
    - complicates fabrication due to different sizes
  - Needed for inbound countermass - not needed in 0g option

- Spin-up/spin-down cycles
  - Mid-course correction problems

- "De-spun" joint for power and communication

g$_a$ Cryo/AB Tether Deployment Scenario

This chart outlines the vehicle deployment scenario. Omitted from this chart, for the interest of simplicity, are mid-course corrections, which would follow the same deployment scenario.

g_a Cryo/AB Tether Deployment Scenario

Reference

- Post TMI, RCS fires to separate MTV (propulsion and AB) and MEV (+ transfer habitat) and deploy tether
  - tether slips freely through crawler
- Crawler clamps to CM point on conductive tether to finish deployment
  - 128m tether length outbound
  - 161m tether length inbound
- RCS fires to accelerate end masses to 4 rpm
- Crawler is positioned at CM and deploys solar array and communications laser
- Post Mars arrival, RCS fires to stop rotation
- Solar array/communications laser retract and crawler moves to MEV
- Tether is reeled in, maintaining slight tension
- RCS fires to slow approach to manageable STET speed
- Post berthing, Mars operations commence
- Reverse scenario after TEI using MTV AB and propellant tanks as countermass to transfer habitat

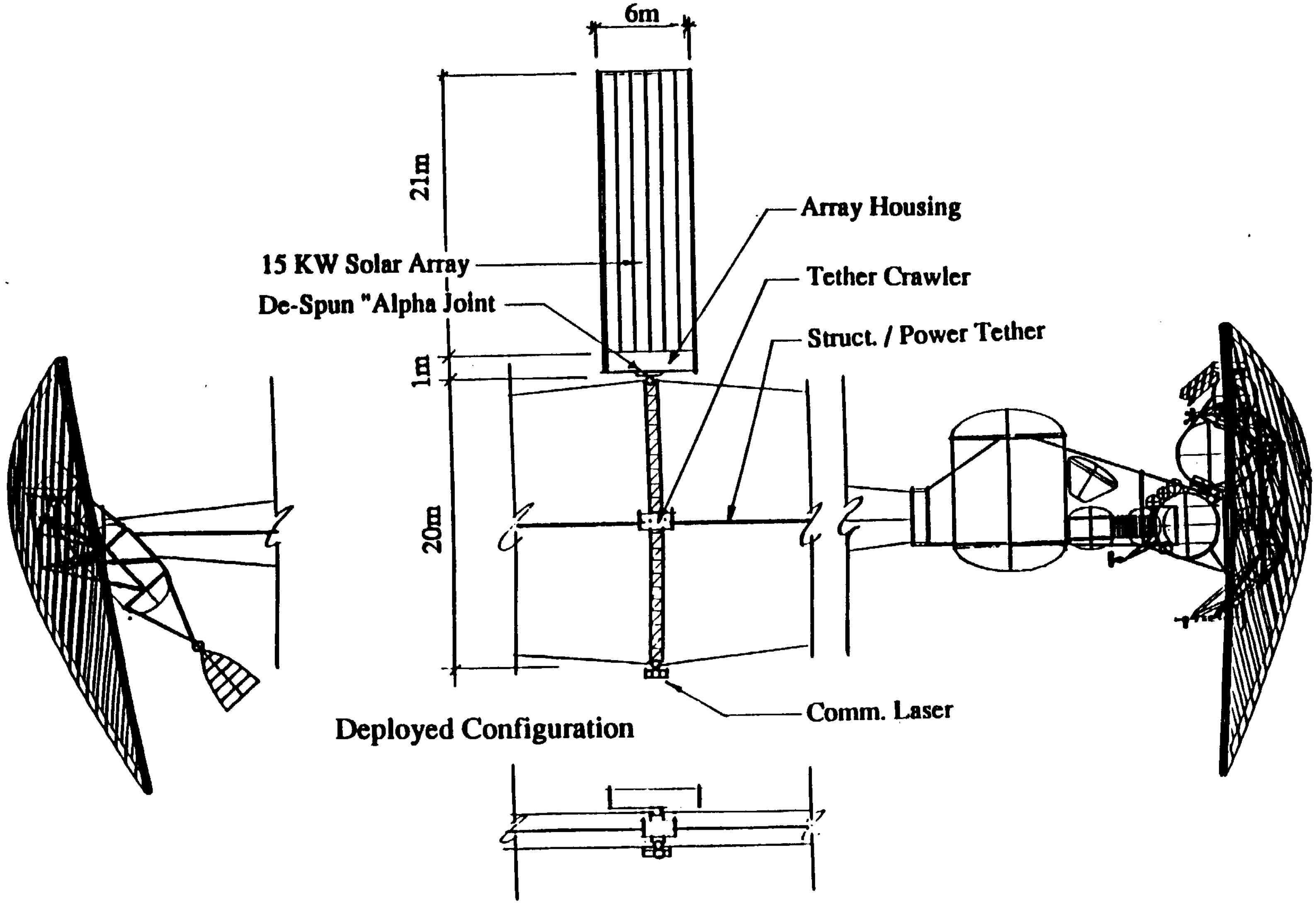
Alternative

- Deploy tether to twice intended length, small ΔV for rotation, then reel tether to nominal length
  - saves propellant, but increases tether mass

g$_a$ Cryo/AB "Crawler/Mast" Configuration

This chart shows a detail of the central "crawler/mast" in the deployed and collapsed configurations. The solar array and the communications laser deploy on a deployable truss to separate the tethers and form a planar "beam". In the collapsed configuration, the solar array and the communications laser fold-up, spin 90° and package below the transfer hab on the MTV.

g_a Cryo/AB "Crawler/Mast" Configuration



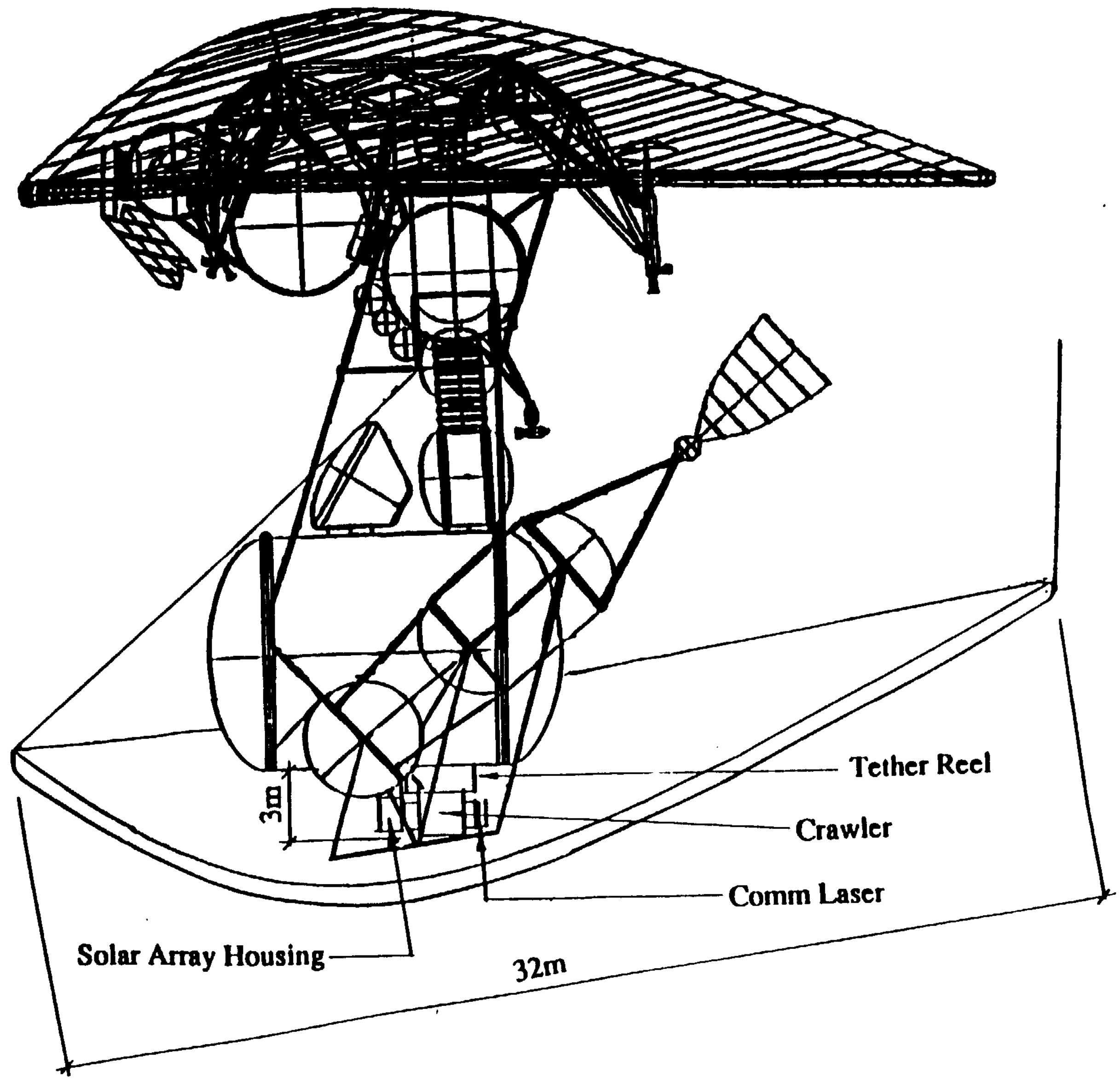
Deployed Configuration

Collapsed Configuration

g_a Cryo/AB Packaging Configuration

This chart shows the packaging configuration for the solar array, communications laser, crawler, and tether reel. Due to aerobraking constraints, the MTV aerobrake has to be 2m larger than the MEV aerobrake, which will cause problems in fabrication commonality. The MTV has been designed so that the transfer module and the artificial gravity equipment can slip out to deploy the tether for spin-up.

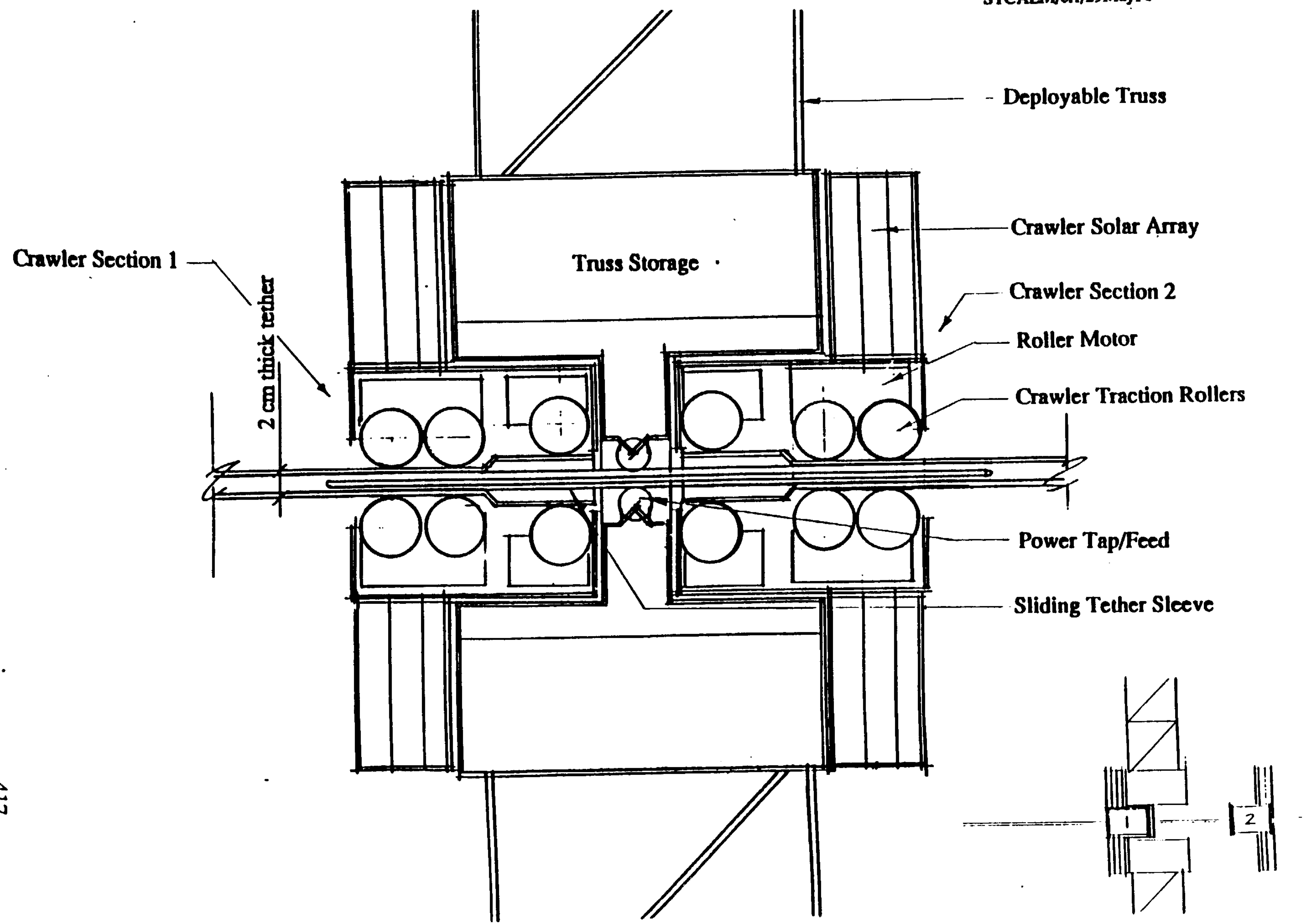
g_a Cryo/AB Packaging Configuration



Tether Crawler Configuration

A detail of the tether crawler is shown on this chart. The crawler is divided into 2 sections so that one section can always be at the CM to support the deployable truss and the tether. The crawler taps into the aluminum conductor to transfer power from the power source to the habitation areas. Each crawler section has 2 small solar arrays for power during movement along the tether and 2 roller motors.

Tether Crawler Configuration



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Conductive Tethers

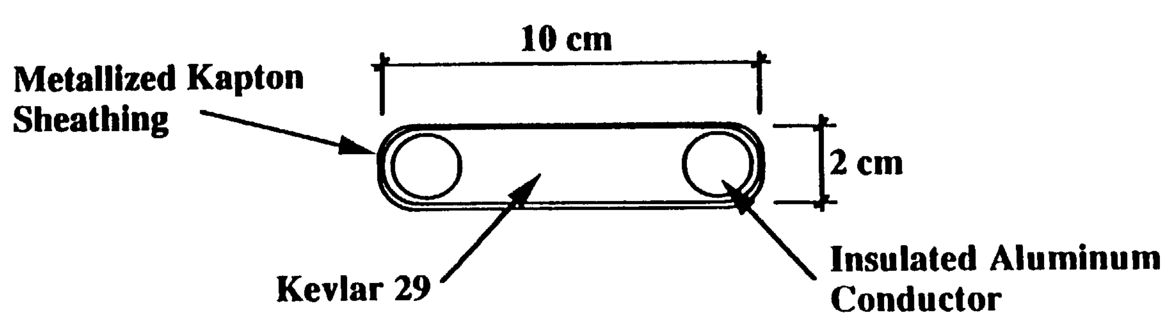
- Transmitting electric power through long flexible cables is standard practice on Earth
  - technology issues well understood
- Maintaining electrical contact between the mobile "crawler" and the tether
  - similar to track lighting, sliprings, electric motors and generators, electric subway trains, and trolleys
  - a Solar Power Satellite concept (1980) incorporated a 5 GWe slipring
  - partially despun spacecraft use lower power sliprings regularly
  - SSF solar arrays will use sliprings to carry tens of kWe
- The Remote Manipulator System on SSF will be much like a tether crawler
  - exception is that it uses power rather than providing it
  - crawls along SSF truss, stopping periodically to plug into electrical outlets
- Technology demonstration in 1991 on Tethered Satellite System (TSS) Shuttle flight
  - conductive tether with plasma contactors for electrodynamic experiment

Conductive Tethers

Conductive tethers have been used in this study to simplify the power transmission method. Conductive tethers are not a simple technology as demonstrated by the examplesgiven on the following chart. Conductive tethers also simplify the realing process because of the reduced number of cables.

Conductive Tether Properties

- Kevlar 29
- 1.2 x 10^6 N stress
- Safety factor = 1.5 (using 3 tethers)
- (3) 180m tethers
  - 161m nominal separation
  - 56m radius to transfer habitat
- "Ribbon" shaped cross section
  - to avoid entanglement during reeling/unreeling cycles
  - easier crawler operations
  - 20 cm^2 cross-sectional area
  - radiates conductor heat better due to increased surface area



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V. Support Systems

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Support Systems for the Mars Cryo/Aerobrake Transfer Vehicle.

The support systems necessary for the Mars Cryo/Aerobrake Transfer Vehicle consist of the interrelated and interdependent tasks of ground, launch, and on-orbit processing. Ground processing tasks for the Mars Vehicle include interface identification and verification as well as integrated systems testing. As the interface diagrams show, each part of the Mars Vehicle is connected (mechanically, electrically, data-wise, and/or fluid-wise) to almost every other part. Earth-to-orbit (ETO) launch processing is constrained by both ground and on-orbit considerations. These tasks include launch site preparation, integrating the payload (in this case, the pieces of the Mars Vehicle) with the Heavy Lift Launch Vehicle (HLLV), and manifesting. The scheduling of hardware to be launched is bounded on one side by the ground test and verification program and on the other side by the on-orbit assembly plan. The selection of Assembly Node and assembly means (robotic, EVA, mix, etc.) are part of this analysis. The systems, facilities, plans, and purposes for each of these three levels of support are included within and represent the magnitude of effort necessary before a Mars vehicle is actually ready to fly..

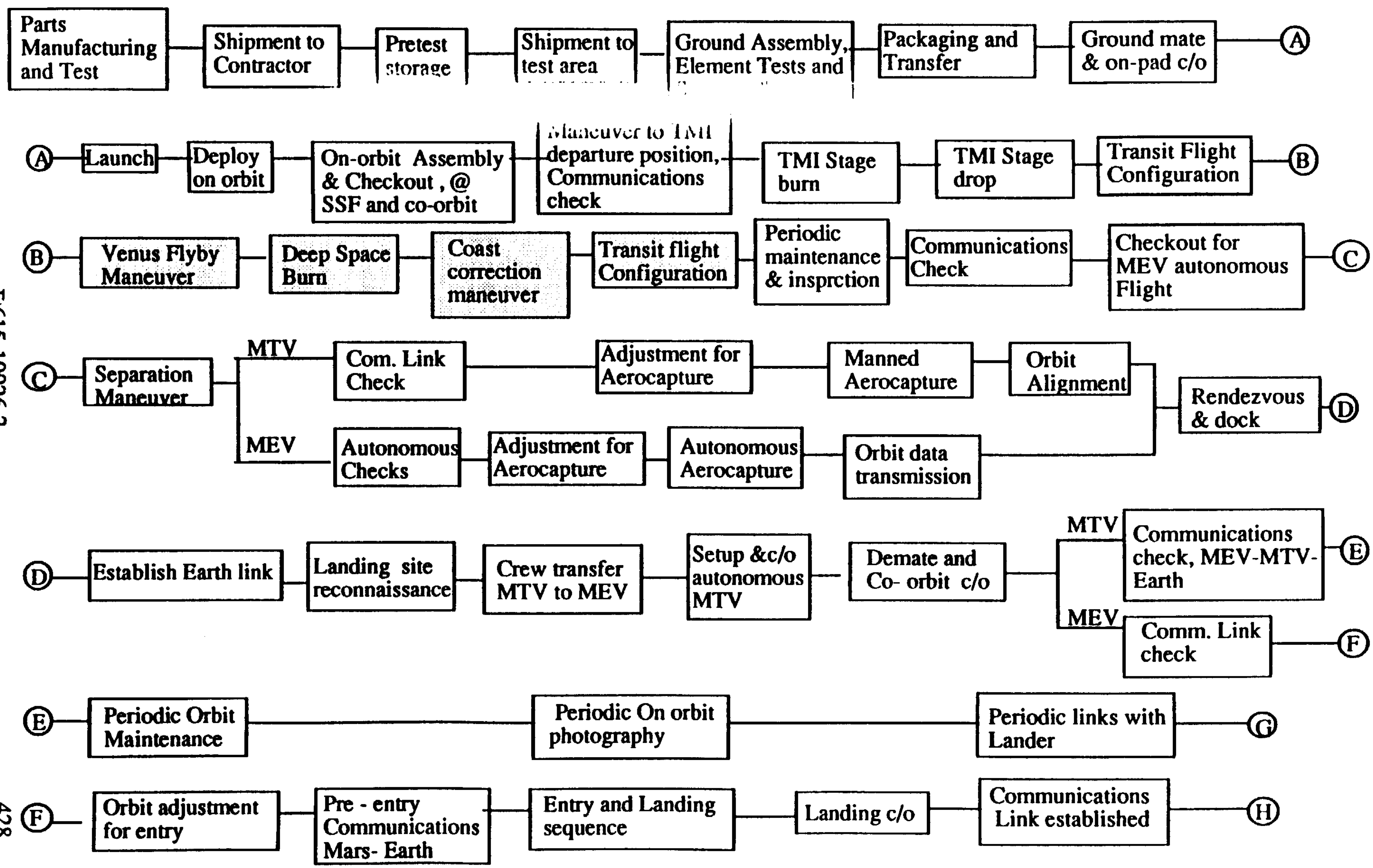
Ground Processing. The first level, ground-based operations, begins with the identification of system interfaces for the Cryo/Aerobrake Vehicle. Subsystem interfaces are to be performed by the manufacturer; however, once complete systems have been delivered to the launch site, it is planned to perform system to system integration in order to test and verify interfaces and system flight readiness. The recommended approach is to use flight hardware to the greatest extent possible during system test and verification. The ground processing flow to accomplish these interface tasks determine when each system(s) must be available and when each will be ready for launch. The generic ground process involves: (1) receiving and inspection of the system(s); (2) assembly of system to system; (3) verification of interfaces and testing for flight readiness; (4) disassembly of system from system; (5) storage of system for other subsequent interface tests; and (5) processing of system for launch.

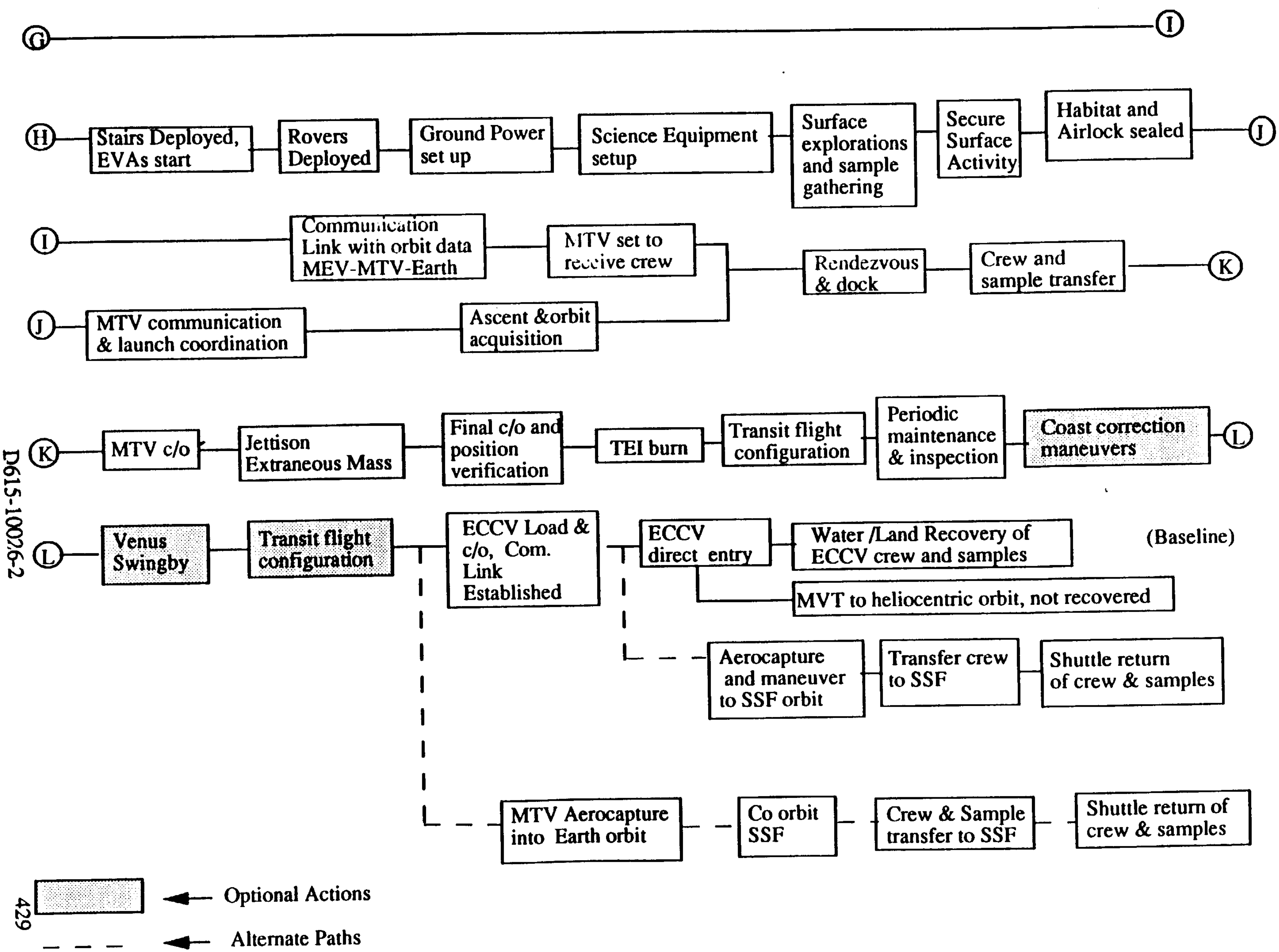
Launch Processing. Launch processing and sequencing constitute the second level of support systems. Processing tasks include integrated assembly and checkout of Mars Vehicle systems with the ETO vehicle. One of the most significant impacts to the assembly and launch facilities as well as to the launch vehicle itself may be the option of launching the aerobrakes fully integrated (the "Ninja Turtle" concept). This concept holds promise for reducing on-orbit assembly problems but raises some processing and launch vehicle compatibility issues. Manifesting analyses are dependent not only upon the ground and on-orbit operations but also upon the selection of the ETO launch vehicle. Several manifesting scenarios have been studied for a variety of HLLVs. In the majority of cases, the limiting factor is found to be payload volume, not mass, capacity.

On-orbit Processing. On-orbit operations, the third level of support systems, pertains to the assembly (and, for reuseable vehicles, the disassembly and refurbishment) of the Mars Cryo/Aerobrake Vehicle. The choice of Assembly Node includes factors such as location, robotic and man-tended capabilities, accessibility, micrometeoroid/debris protection, operating systems, and on-orbit storage. An on-orbit assembly analysis has been performed for the reference vehicle (with the added constraint that the aerobrakes must be assembled in space) based upon one possible assembly platform which may be suitable for the Cryo/Aerobrake vehicle. This platform was designed to solve two of the major problems with assembly of the vehicle in Low Earth Orbit (LEO): debris protection and aerobrake construction. The STS External Tanks serve as both protection and a base upon which assembly mechanisms, storage, and vehicle integration may be performed. This is not intended to be the final solution to these problems; rather, this study serves to show one possible solution at one possible node. The resulting analysis indicates that the main delimiter in assembly time is the launch frequency of the ETO vehicle.

Space

Mars Mission Operational Task Flow
Cryo/Aerobrake

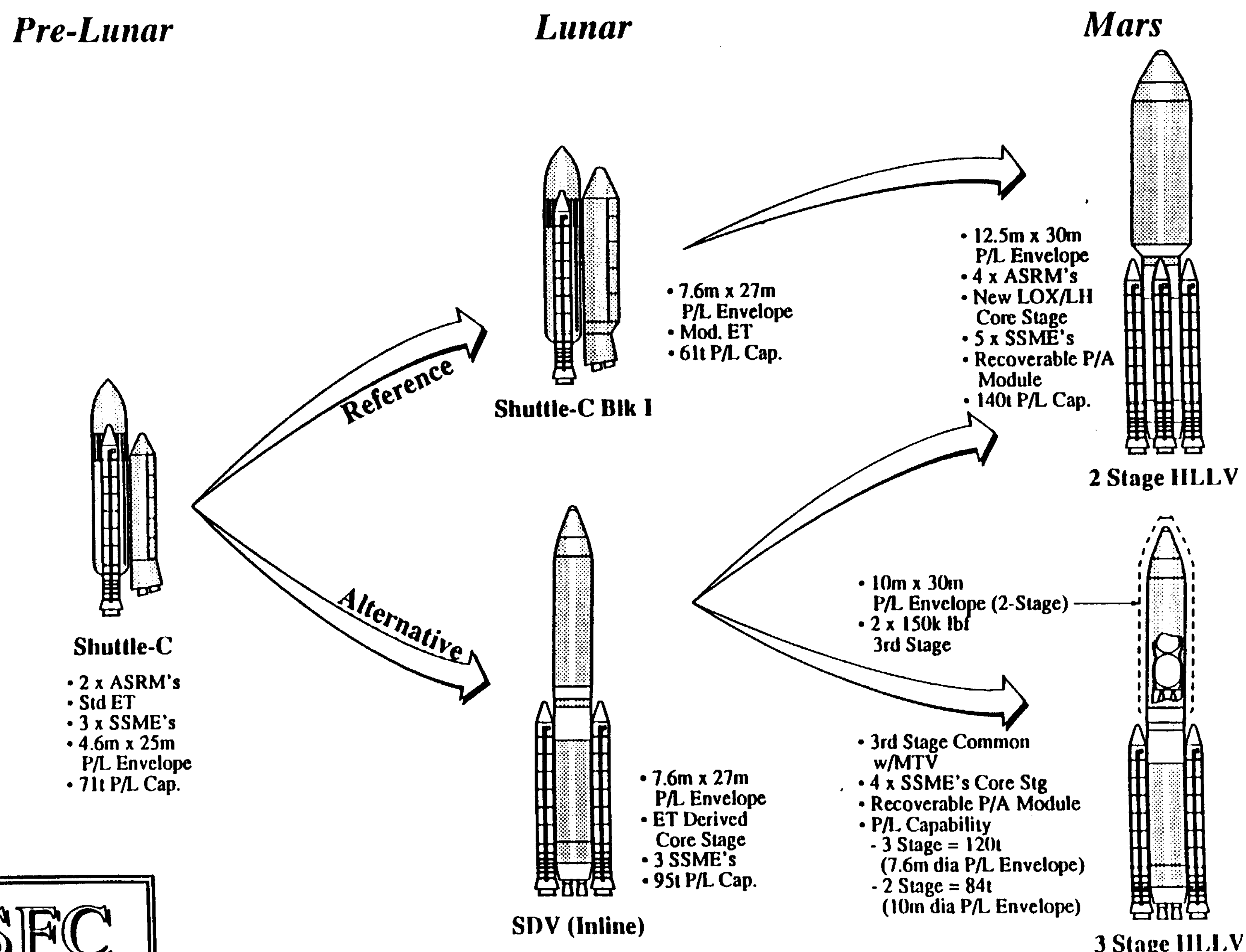



G
I
H
Stairs Deployed, EVAs start
Rovers Deployed
Ground Power set up
Science Equipment setup
Surface explorations and sample gathering
Secure Surface Activity
Habitat and Airlock sealed
J
I
Communication Link with orbit data MEV-MTV-Earth
MTV set to receive crew
Rendezvous & dock
Crew and sample transfer
K
J
MTV communication & launch coordination
Ascent &orbit acquisition
K
MTV c/o
Jettison Extraneous Mass
Final c/o and position verification
TEI burn
Transit flight configuration
Periodic maintenance & inspection
Coast correction maneuvers
L
L
Venus Swingby
Transit flight configuration
ECCV Load & c/o, Com. Link Established
ECCV direct entry
Water /Land Recovery of ECCV crew and samples
(Baseline)
MVT to heliocentric orbit, not recovered
Aerocapture and maneuver to SSF orbit
Transfer crew to SSF
Shuttle return of crew & samples
MTV Aerocapture into Earth orbit
Co orbit SSF
Crew & Sample transfer to SSF
Shuttle return of crew & samples
Optional Actions
Alternate Paths

Shuttle Derived Launch Vehicle Approach For Lunar/Mars Initiative

This is a MSFC chart showing the launch vehicles considered in Earth-to-Orbit launches. We have done manifesting scenarios for the reference line of vehicles. These scenarios are shown in the following two charts, indicating what is manifested, type of vehicle, the number of launches, and the estimated payload mass per launch for the first three missions.

Shuttle Derived Launch Vehicle Approach for Lunar/Mars Initiative



MSFC

Mars Mission Vehicle Manifests by Year
Shuttle C/Z

	Flight #	1	2	3	4	5	6	7- 12
2015 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle Z	Shuttle C	Shuttle Z	Shuttle Z	Shuttle Z
	Manifest	MTV aerobrake, assembly equipment	Crew systems, ECCV, structure, comsats, consumables	TEIS O2 propellant, tankage	TEIS H2 propellant, tankage & engines	MEV aerobrake, assembly equipment	Ascent vehicle , descent vehicle, surface payload	MTV - Earth departure propellant & engines
	Mass	41.0 t	50t	91t	15t	12t	75t	91t each launch

	Flight #	1	2	3	4	5	6 - 9
2017 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle Z	Shuttle C	Shuttle C	Shuttle Z
	Manifest	MEV 1 aerobrake, assembly equipment	50t payload, descent stage for MEV1	MEV 2 aerobrake, assembly equipment	50t payload, descent stage for MEV 2	MEV1-2 & TMIS inter -connect structure, Nav kit	TMIS engines and departure propellant
	Mass	12 t	70t	12t	70t	20t	91t each launch

	Flight #	1	2	3	4	5	6-11
2018 Mars Departure	Launch vehicle	Shuttle Z	Shuttle C	Shuttle C	Shuttle Z	Shuttle Z	Shuttle Z
	Manifest	MTV aerobrake, assembly equipment	Crew Systems ECCV, structure, consumables	TEIS propellant tankage O2 & H2	MEV aerobrake, assembly equipment	Ascent vehicle descent vehicle, surface payload	MTV- Earth departure propellant & engines
	Mass	41t	50t	69 t	12t	75 t	85 t each launch

ADVANCED CIVIL SPACE SYSTEMS

Mars Mission Manifests- Cryo/aerobrake and NTR 140 t HLLV

BOEING

STCAEM/PB/9,15,90

HLLV : Shroud Size- 30 meters x 10 m dia., 140 t throw weight

Cryo/Aerobrake

Flight #	1	2	3	4	5	6	7	8
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MEV Aerobrake (Ninja Turtle), Descent stage, surface cargo, assembly equipment, consumables	Ascent stage, interconnect structure (MEV-MTV), TEI departure stage	MTV Hab Module, ECCV, MTV Aerobrake (Ninja Turtle), interconnect structure (MTV-TMIS), assembly equipment, consumables	TMIS tankage, Engine set, Structure, assembly equipment, Top-off equipment	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant	TMIS tankage and propellant (wet tanks and top-off)
Mass	67.8 t	138.7 t	100 t	140 t	119.7 t	119.7 t	119.7 t	140 t

NTR

Flight #	1	2	3	4	5	6	7
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MEV Aerobrake (Ninja Turtle), MTV Hab, Ascent module, Descent module, Main truss, assembly equipment	Mars Departure Structure, Surface payload, Mars Departure Tankage	TMI Tank	TMI Tank	Engine, Shield structure, Mars Arrival/ Departure Tank	Reactor Engine and Shield, Mars Arrival/ Departure Tank	Propellant for top-off
Mass	132.8 t	121.1	140 t	140 t	85.3 t	94.3 t	26.6 t

Mars Mission Manifests-
SEP and NEP
140 t HLLV

STCAEM/PB/9,15,90

HLLV : Shroud Size- 30 meters x 10 m dia., 140 t throw weight

SEP

Flight #	1	2	3	4	5
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MTV Hab, Array Deployment Mech., Communications, ACS, Avionics, PPUs, Power Distribution & Control, Main Truss, Array Structure, Experiment Platforms, Assembly Equipment	Transfer Arrays (18 of 18), MEV Aerobrake (Ninja Turtle), Assembly Equipment	Descent Module, Ascent Module, Surface Payload, Main Truss	Propellant Tanks, Propellant, Radiators, Array Blankets (14 of 18)	Propellant tanks, Propellant, Thruster pods, Array Blankets (4 of 18)
Mass	82.2 t	45.5 t	88.8 t	104.3 t	96.5 t

NEP

Flight #	1	2	3	4	5
Launch Vehicle	HLLV	HLLV	HLLV	HLLV	HLLV
Manifest	MTV Hab, MEV aerobrake (Ninja Turtle), ACS, Power conditioners, Communications, Assembly Equipment	Descent Module, Ascent Module, Surface Payload, Auxiliary Radiator, Power distribution and control	2 Propellant tanks, 3 thruster pods	1 Main Radiator, 1 thruster pod, 3 propellant tanks, 1 Auxiliary Radiator	2 nd Main Radiator, Reactors, Shields, Turbo pumps, Misc.
Mass	85.1 t	118.1 t	115.1 t	132.6 t	104.3 t

Mars Mission Manifests- Cryo/aerobrake and NTR 84 -120 t HLLV

STCAEM/PB/9,15,90

HLLV 1 : Shroud Size- 30 meters x 10 m dia., 84 t throw weight
HLLV 2 : Shroud Size - 30 meters x 7.6 m dia., 120 t throw weight

Cryo/Aerobrake

Flight #	1	2	3	4	5	6	7-10	11
Launch Vehicle	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 2	HLLV 2	HLLV 1
Manifest	MTV Hab Module, Surface Payload, Assembly Equipment	MEV Aerobrake (Ninja Turtle) Habitat refurbish/ consumables, Assembly Equipment	Descent Module, Ascent Module, structure	MTV Aerobrake (Ninja Turtle), Habitat refurbish/ consumables, Assembly Equipment, ECCV, TEI Tanks & Engines	TEIS Propellant, consumables	TMI Propellant & Engines	TMI Propellant and tanks	Top-off Propellant and equipment, Habitat refurbish/ consumables
Mass	78.8 t	40.1 t	49.1 t	76.4 t	84 t	117.7 t	114.5 t	84 t

NTR

Flight #	1	2	3	4	5	6	7	8	9
Launch Vehicle	HLLV 2	HLLV 1	HLLV 1	HLLV 1	HLLV1	HLLV 1	HLLV 1	HLLV 1 or 2	HLLV 1 or 2
Manifest	Descent Module, Ascent Module, Surface Payload, MEV aerobrake (Ninja Turtle), Assembly Equipment	Mars Departure/ Earth Arrival Structure, Main Truss, MTV Hab	Engine/ Shield Structure, Mars Departure/ Earth Arrival tank (13.5 t off-loaded),	Mars Arrival/ Departure Tank (1 of 2)	Mars Arrival/ Departure Tank (2nd of 2)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Earth Departure Tank (1 of 2, 69.3 t off-loaded)	Off-loaded Propellant from Flight 3 &6	Off-loaded Propellant from Flight 6 & 7, Reactor Engine and Shield
Mass	112.2 t	46.6 t	84.0 t	82.9 t	82.9 t	84 t	84 t	79.5 t	84 t

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Mars Mission Manifests-
SEP and NEP
84 - 120 t HLLV

ADVANCED CIVIL SPACE SYSTEMS

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HLLV 1 : Shroud Size- 30 meters x 10 m dia., 84 t throw weight
HLLV 2 : Shroud Size - 30 meters x 7.6 m dia., 120 t throw weight

SEP

Flight #	1	2	3	4	5	6
Launch Vehicle	HLLV 2	HLLV 1	HLLV1	HLLV1	HLLV 1	HLLV1
Manifests	Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment	Array Deployment Mech., Communications, ACS PPUs, Avionics, Main Truss (3 of 4), Power Distribution & Control, Array Structure, Experiment Platforms	Propellant & Tanks (1 of 4), Radiators (2 of 2), Main Truss (4th of 4), Transfer Arrays (10 of 18)	Propellant & Tanks (2nd of 4), Transfer Arrays (8 of 18), Thruster Pods (2 of 2)	Propellant & Tanks (3rd of4), Array Blankets (15 of 18)	Propellant & Tanks (4th of 4), Array Blankets (last 3) MTV Hab.
Mass	112.2 t	34.9 t	62.9 t	63.8 t	60.3 t	83.2 t

NEP

Flight #	1	2	3	4	5	6	7
Launch Vehicle	HLLV 2	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV 1	HLLV 2
Manifests	Descent Module, Ascent Module, Surface Payload, MEV Aerobrake (Ninja Turtle), Assembly Equipment	Power Distribution and Control, Structure, ACS, Power Conditioning, Communications, Avionics	Propellant and Tanks (1 of 5), Thruster pods (3 0f 4)	Main Cycle Radiators (2 of 2), Thruster pod, Propellant and Tanks (2nd of 5)	Auxiliary Radiators (2 of 2), Turbopumps/ Thermal loops,	Propellant and Tanks (3rd of 5), MTV Hab,	Reactors & Shields, Propellant and Tanks (last 2)
Mass	112.2 t	48 t	78.1 t	61.4 t	76.6 t	77.3 t	101.4 t

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HLLV Optional Manifesting

Optional Manifesting of the four vehicle options was developed for a medium and large class HLLV. The analysis was completed by using theoretical volumetric and total mass calculations.

STCAEM/dks/12June90

HLLV Optional Manifesting

- Manifesting data will be generated by volumetric and mass total calculations
- Aerobrake(s) will be assembled on-orbit
- Deployable truss type mechanisms are feasible
- Manifesting assumed on-orbit assembly at LEO

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On-Orbit Assembly Analysis
(HLLV Missions to LEO)

Proplusion Option / ETO Size	10 meter Dia. 30 meter long 84-120 mT class	12.5 meter Dia. 30 meter long 140-160 mT class	13.75 meter Dia. 38 meter long/22 meter nose cone 200-250 mT class
Cryo/Aerobrake (opposition class)	11 missions 73 mT average	8 missions 101 mT average	5 missions 162 mT average
Solar Electric Power (opposition class)	TBD	5 missions 87.2 mT average	2 missions 218 mT average
Nuclear Electric Power (opposition class)	TBD	5 missions 108.9 mT average	3 missions 181.5 mT average
Nuclear Thermal Rocket (opposition class)	TBD	6 missions 122 mT average	4 missions 183 mT average

'STCAEM/Ad/5 June 90

Requirements For Earth Orbit Support Facility

This is a listing of the groundrules and assumptions used to begin analyzing the sequencing and operations for an orbit assembly facility.

note: this is a point design study

Requirements for Earth Orbit Support Facility

Groundrules
- Multiple (ETO) flights will be used in assembly
- Line of sight communications are to be used
- Extensive use of robotic and telepresent systems will be made
- Minimal EVA activities

Assumptions
- On orbit propellant fueling or launched wet propellant tanks may be used
- Two RMS systems will be used in assembly
- On orbit spares will be 15% of vehicle active component weight and 20% of inactive weight
- Robotic software and sensors will allow supervisory human control
- Proximity operations will be viewed directly or by video
- Assembly schedule will be two (2) years or less

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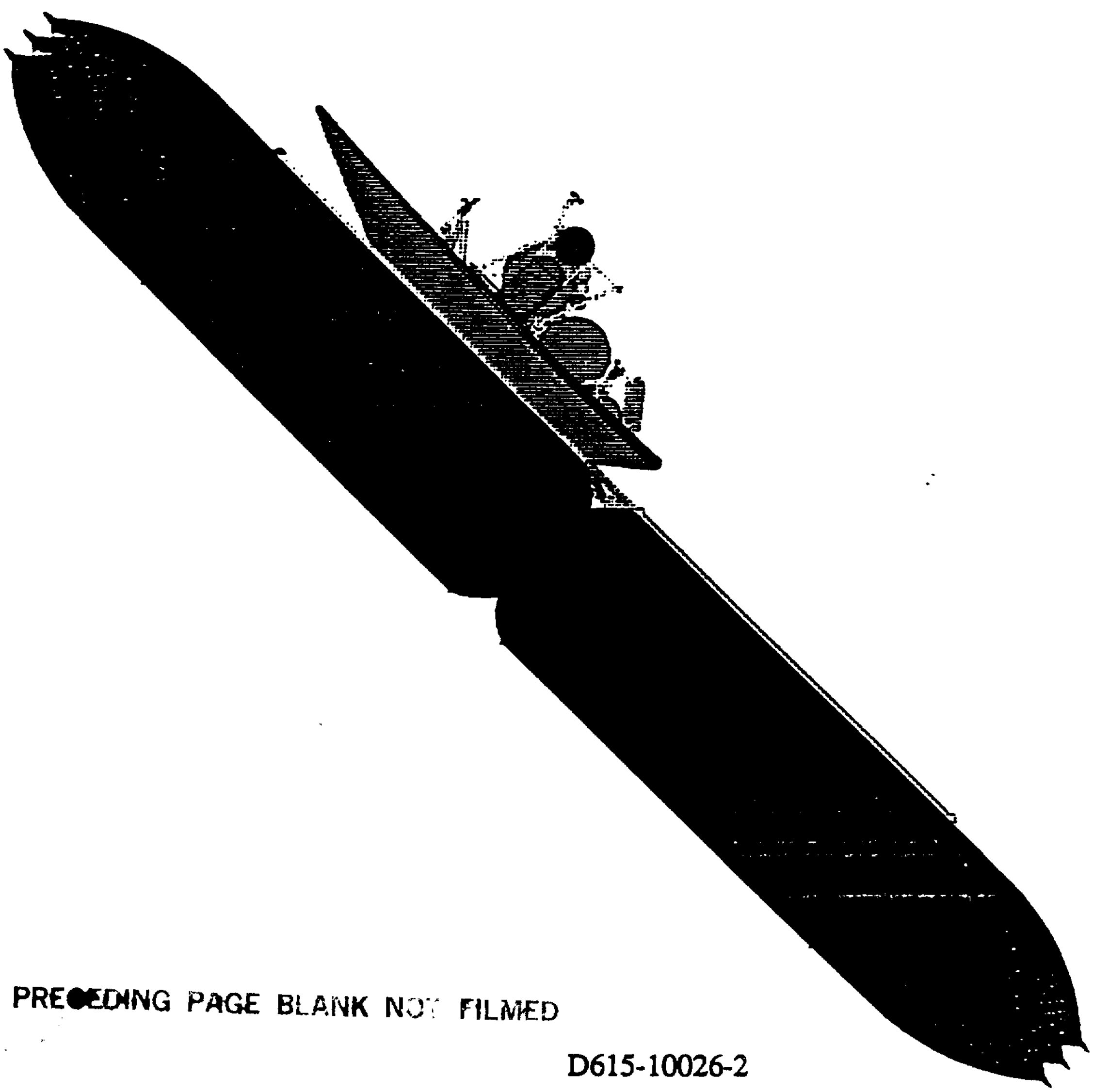
Support Requirements and Concepts Orbital and Space Based Requirements Summary - February 23, 1990



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On-Orbit Assembly

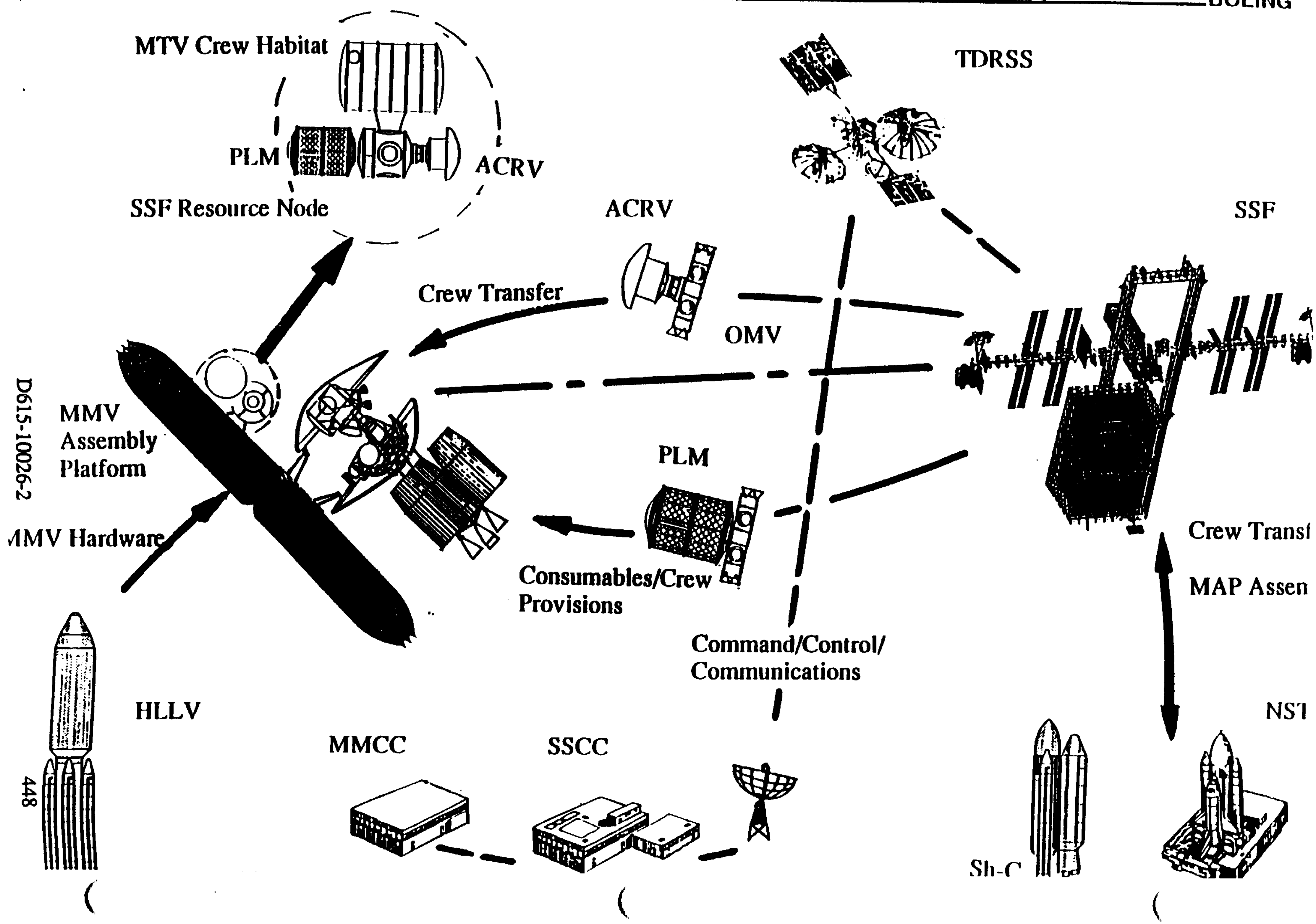
- **Purpose**
  Define orbital and space-based support equipment, crew and facilities requirement/interfaces. By transportation element, for each scenario (Task 5-2).

- **Man Mars Vehicle Baseline**
  - Mars Excursion Vehicle
    - Aerobrake
    - Descent System
    - Ascent System
    - Mars Surface Payload
    - Mars Science Payload
  - Mars Transfer Vehicle
    - Aerobrake
    - Trans Earth Injection System
    - Habitat Module
  - Trans Mars Injection System
    - Core Stack
    - Propellant Tank Set (3 Tanks Baseline)

Groundrules/Assumptions

- Off-SSF assembly of MMV
- HLLV available for MMV launch
  - 4 flights per year
  - On-orbit stationkeeping (< 1 week)
- SSF-based OMV capable of maneuvering complete MMV subassemblies (i.e. MTV crew habitat)
- Maximize automation and robotics for assembly tasks
- MMV LEO departure date---Feb 2016
- MMV has high level of BIT/BITE

MMV On-Orbit Assembly Operations Baseline



On-Orbit Assembly Baseline

MMV assembly at ET-derived MMV Assembly Platform (MAP)
- Constructed prior to MMV FEL
- MAP is self-supporting with power, control, debris protection capability

Crew required for internal subsystem checkout, critical inspections, contingency, repair

All MMV components have standard STS grapple fixtures

RMS capability at MAP
- PRMS---2 MAP mounted 30m arms
- RAMS---4-30m arms (2 on each aerobrake)
- PAS---2 DOF anchors to hold large subassemblies
- ASF---Fixed anchors to store components prior to assembly

MMV aerobrake TPS installed on ground except around field joints
- TPS around field joints installed by PRMS

MAP has line-of-sight communications with SSF

Crew accomodations
- MTV habitat module provides early crew quarters
- Crew transferred from SSF in ACRV/OMV when needed
- SSF resource node contains workstation for MAP local control
- SSF PLM/OMV used for resupply of consumables/crew provisions/MMV spares

Definitions

MAP---MMV Assembly Platform
MMV---Mars Mission Vehicle
SSF---Space Station Freedom
HLLV---Heavy Lift Launch Vehicle
OMV---Orbital Maneuvering Vehicle
ET---External Tank
FEL---First Element Launch
RMS---Remote Manipulator System
PRMS---Platform Remote Manipulator System
RAMS---Remote Aerobrake Manipulator System
PAS---Platform Anchor System
ASF---Assembly Storage Fixtures
SSCC---Space Station Control Center
MMCC---Mars Mission Control Center
STS---Space Transportation System
TDRSS---Tracking and Data Relay Satellite System

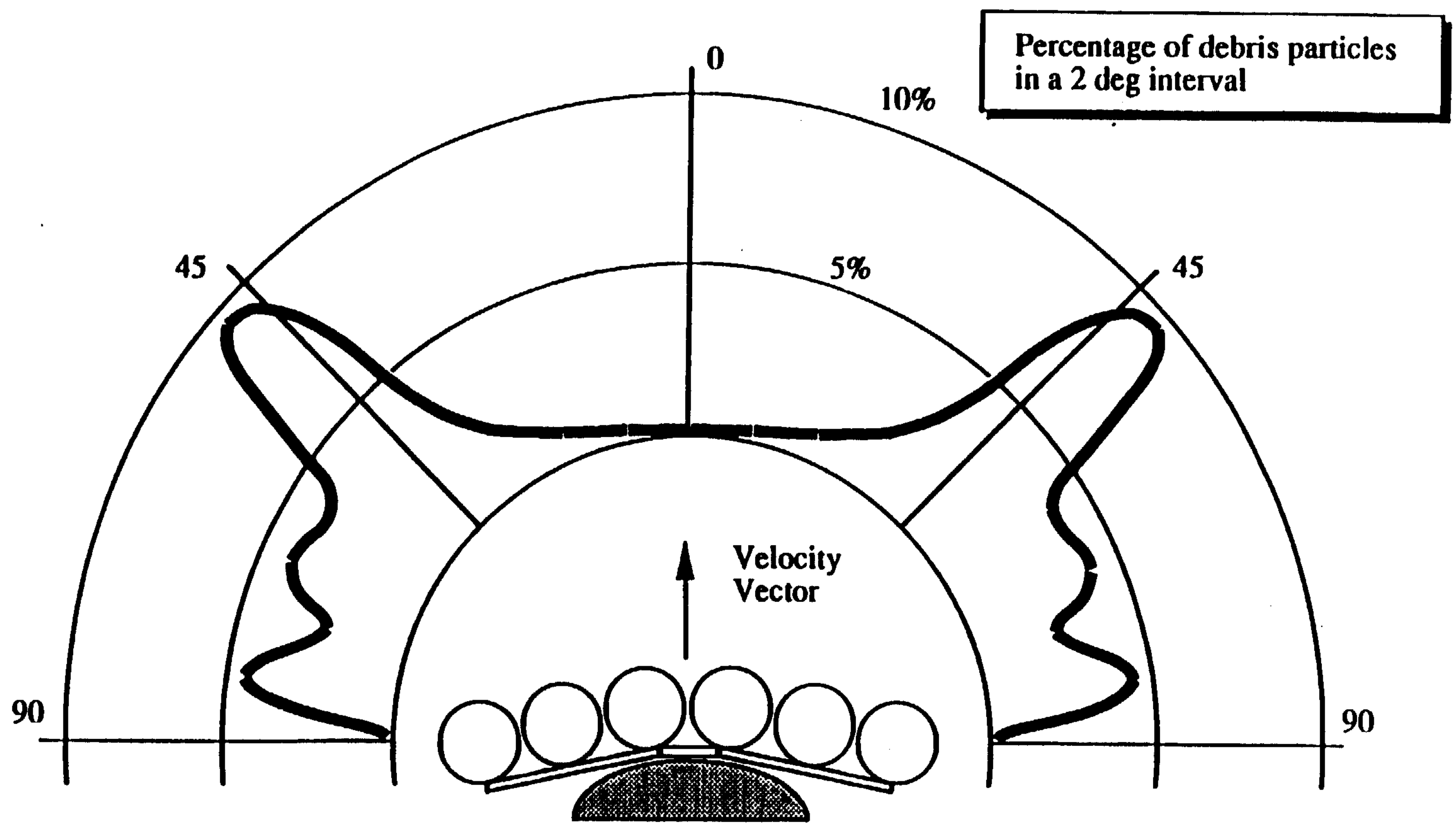
Orbital Debris Environment

- MMV debris environment will be 20-25 times worse than current SSF requirement
  - Orbital debris environment for SSF dated 1985
  - New environment dated April 1989---4-5 times greater flux
  - Later MMV launch date (2016)---5 times greater flux
  - Shorter MMV stay time (10yr/2yr)---reduced probability of impact
  - Will by further modified by LDEF data

- SSF requirement
  - 0.9955 probability of no penetration for each module for 10 years

- SSF debris shielding planned
  - 0.05 in Aluminum shield
  - 4.3 in spacing
  - 0.125 in Aluminum pressure wall

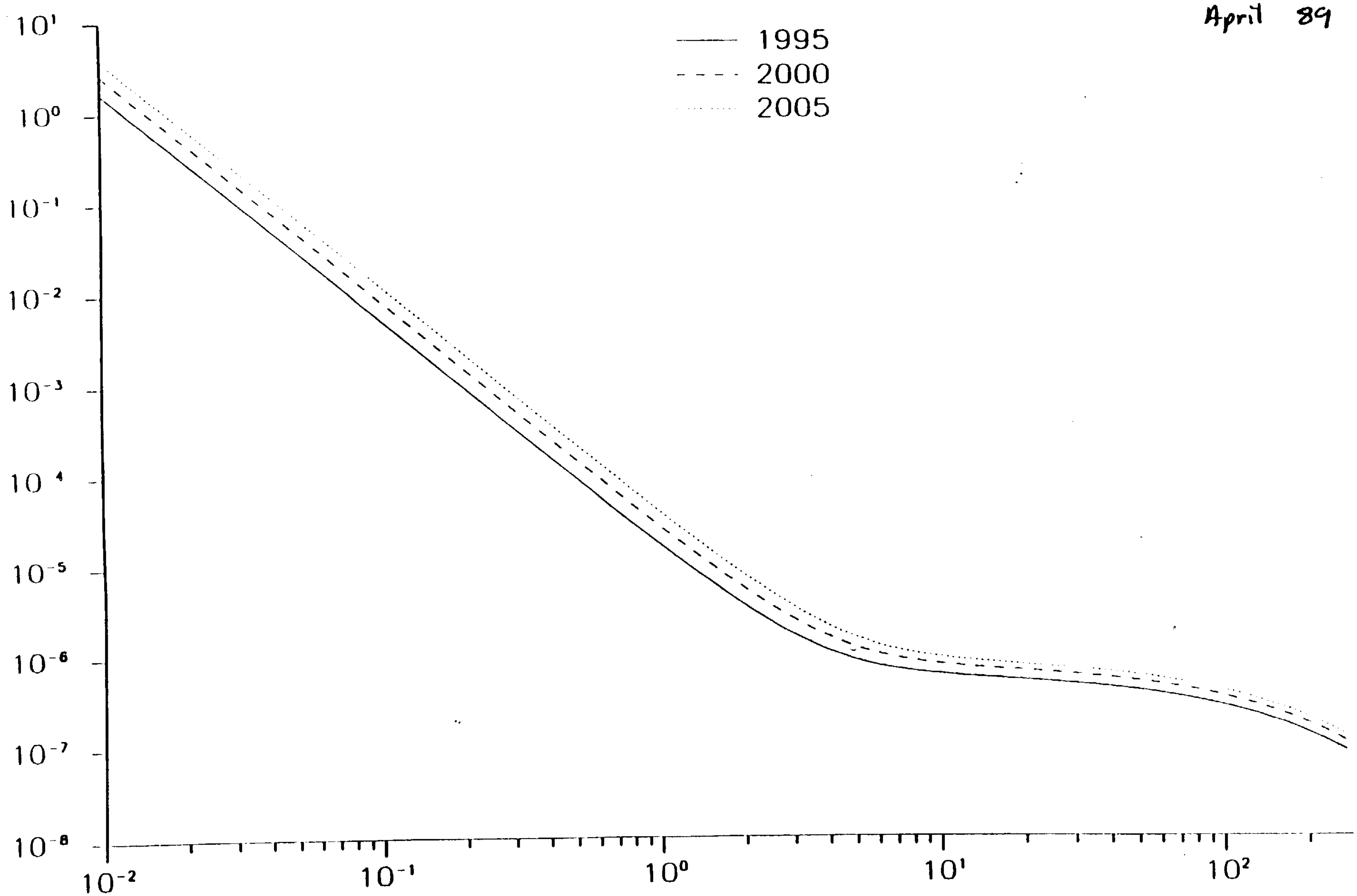
Orbital Debris Environment (Cont)



Normalized Closing Angle Density Function
500 km altitude, average 1990's environment

Change Request Cumulative Flux Distribution
500 km altitude

NASA-TMX
April 89



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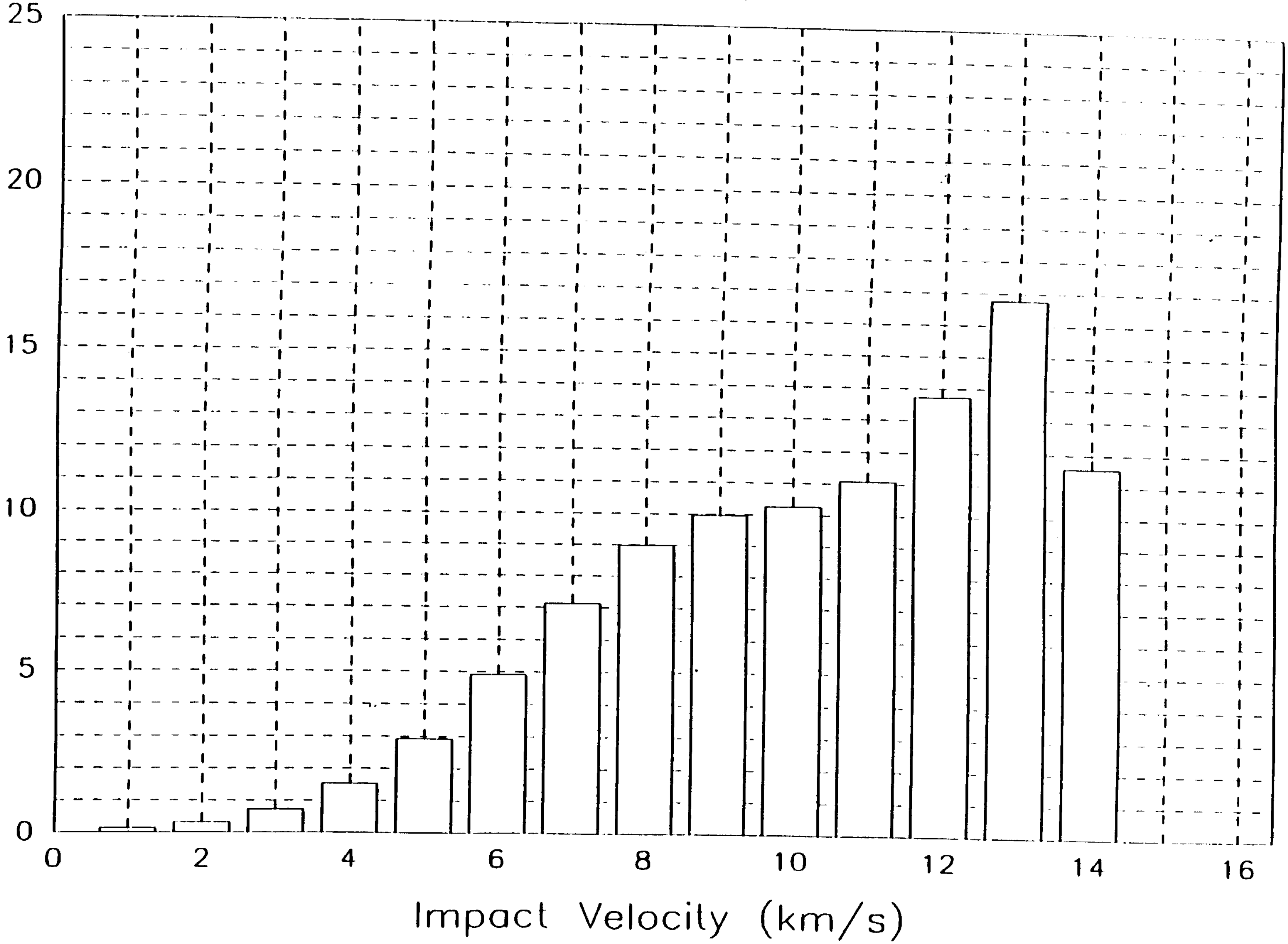
Normalized Closing Velocity Density Function

Change Request



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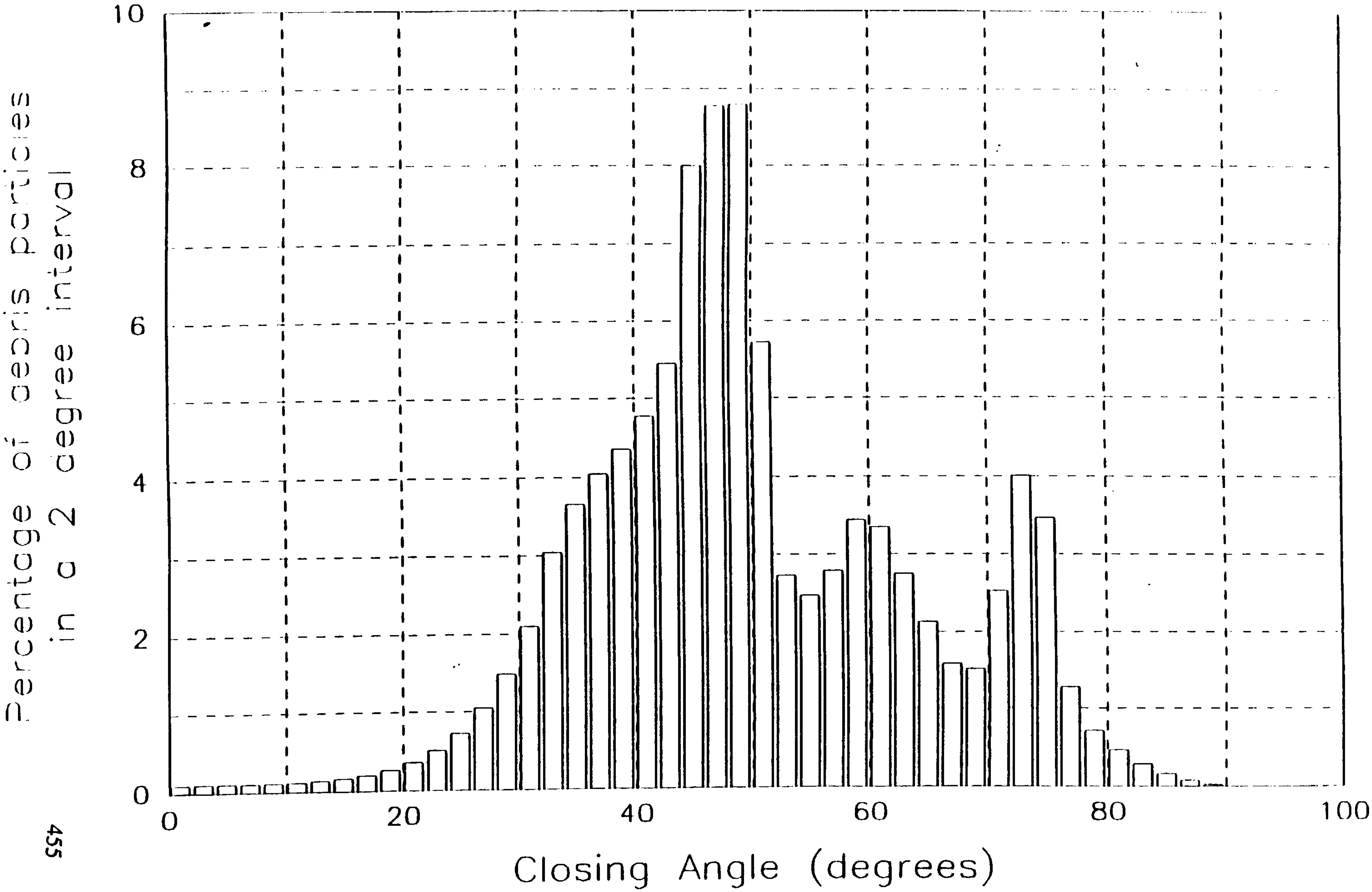
Normalized Closing Angle Density Function
500 km altitude, average 1990s environment



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SSP 30425 REV B

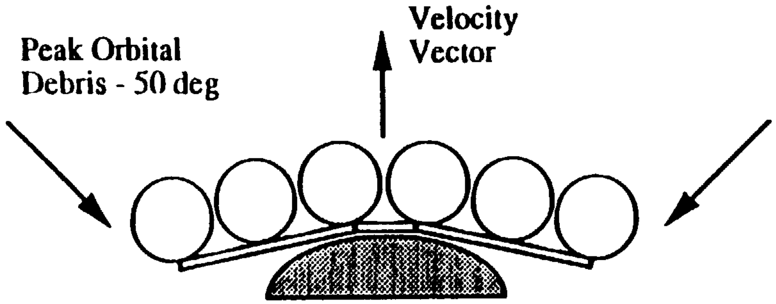
Off-SSF Assembly Node

Assembly Platform vs Integral MMV Assembly

	Advantages	Disadvantages
Assembly Platform	Available prior to MMV FEL Large debris protected area Platform for mounting power, control, communications subsystems Space for parallel assembly tasks/temporary storage of MMV components Platform for additional RMS	Separate vehicle to control (ground/ proximity operations) Additional launches required
Integral MMV Assembly	All required subsystems already available in some form Allows thorough checkout of subsystems prior to launch	Flight hardware used for debris protection Modification of flight power, control subsystems to control vehicle during assembly phase Requires storage space at SSF

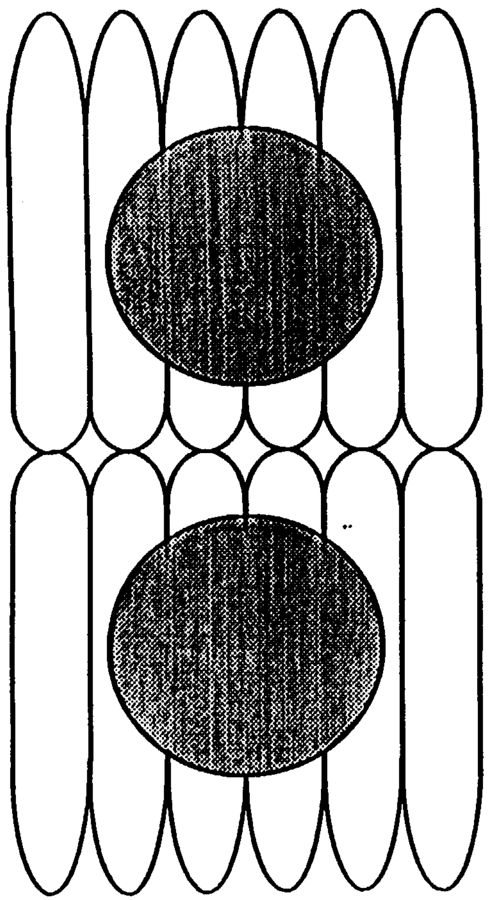
Assembly platform provides large protected assembly workspace with minimum impact to MMV flight hardware

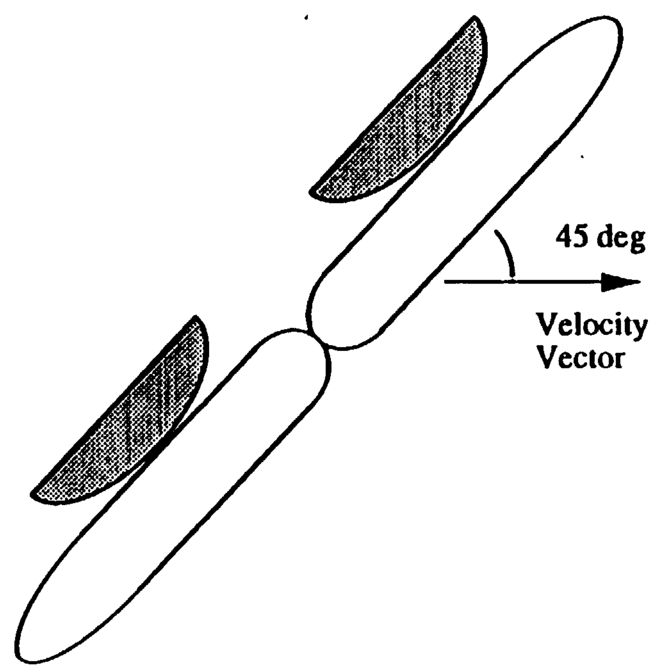
ET Debris Shield Concept for MMV Assembly



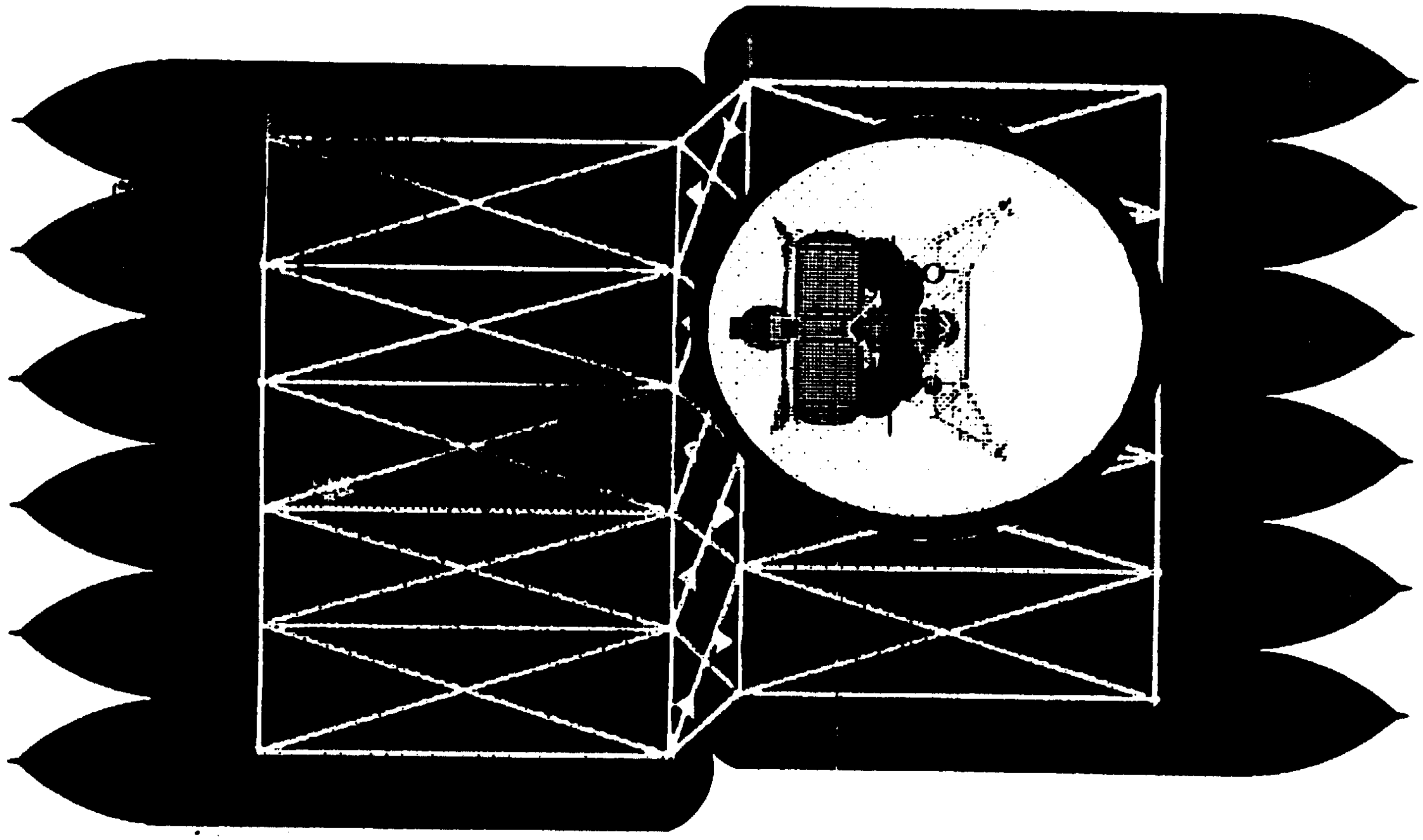
External Tank
47 x 8.4m dia
"Raft" of 6 ET's provides a 47 x 50.4m shield
"Raft" of 12 ET's provides a 94 x 50.4m shield

MTV/MEV Aerobrakes
30 x 27.4 x 7m thick





On-C.bit Assembly



Man Mars Vehicle Assembly Platform (MAP)
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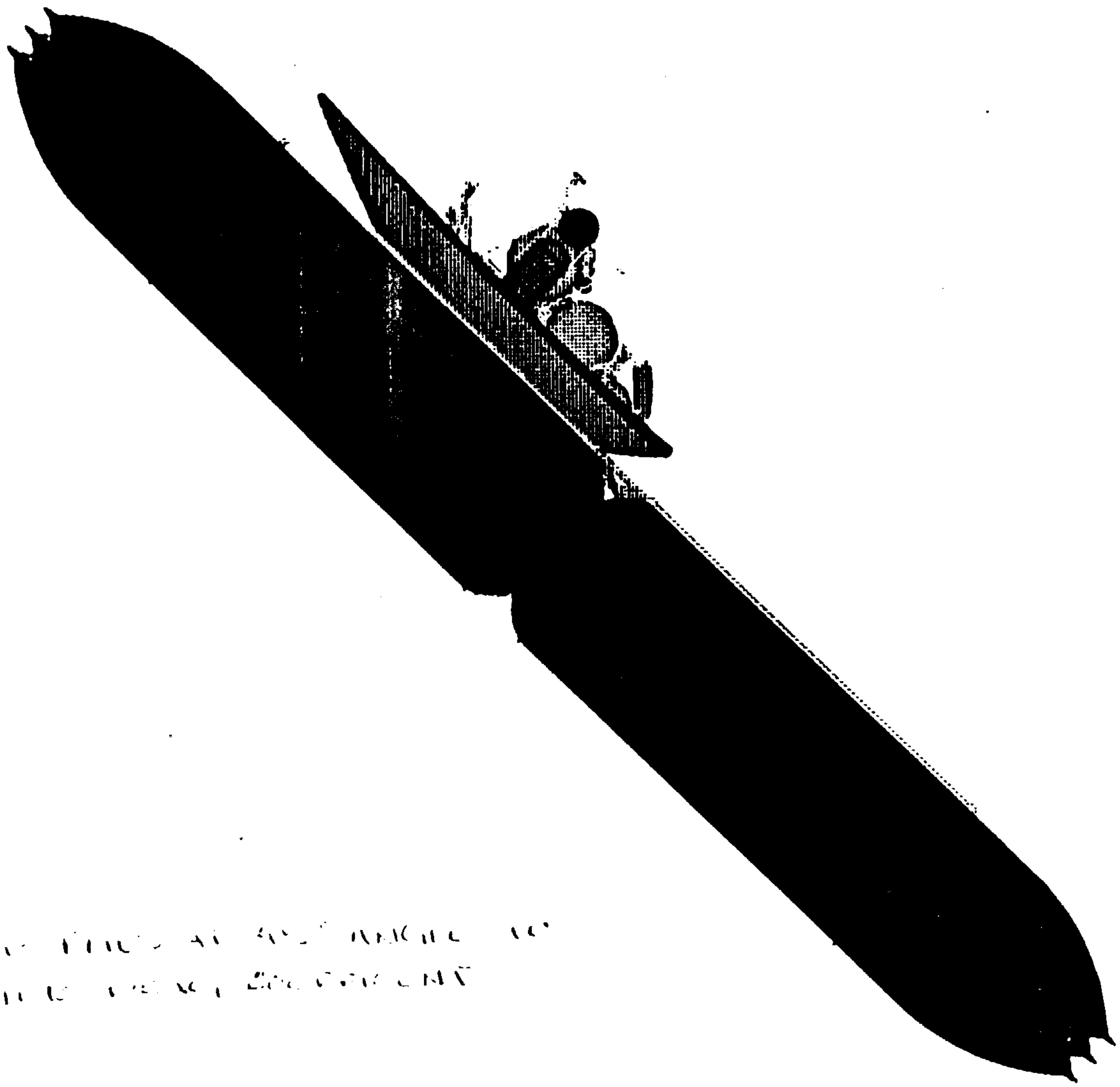
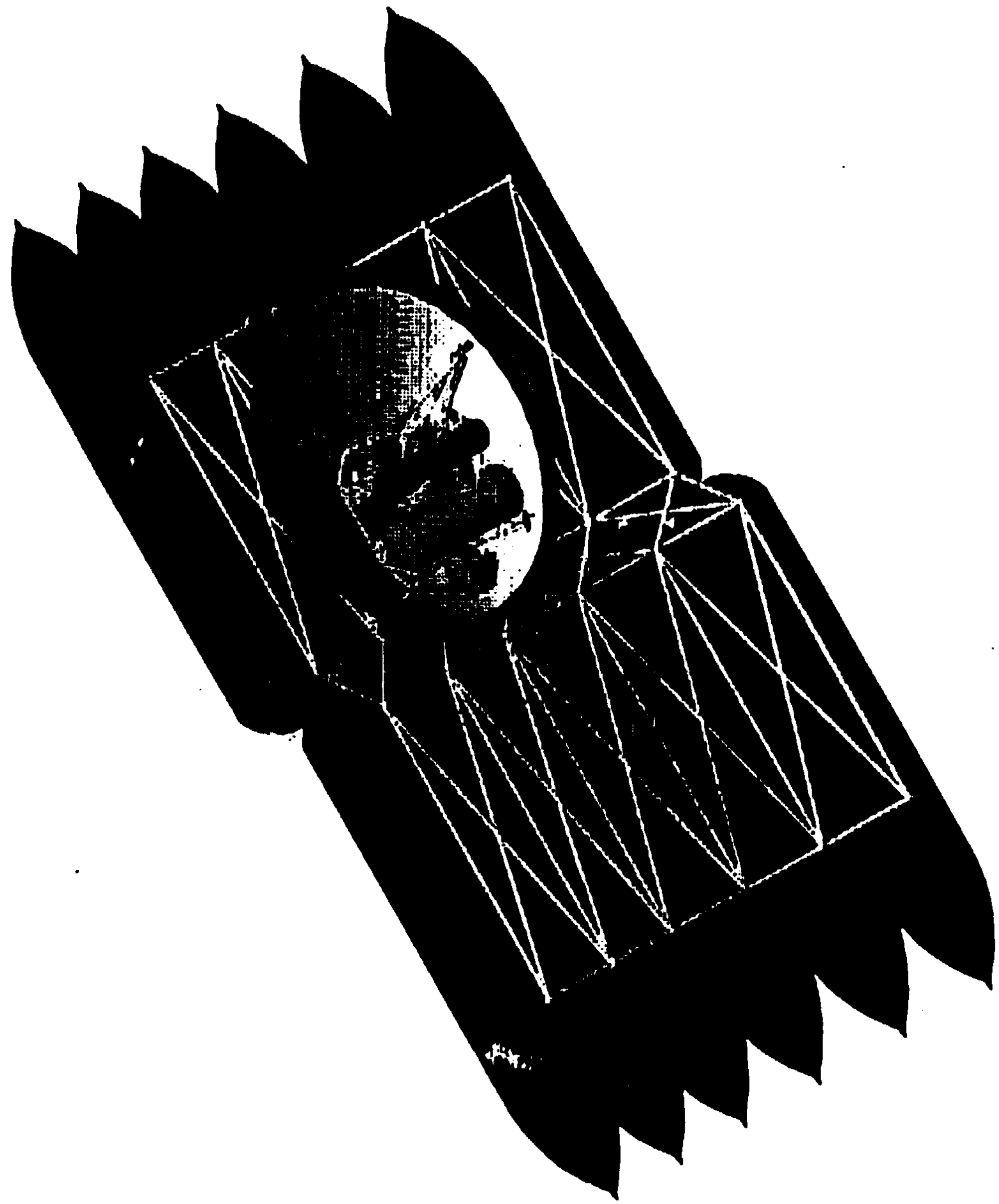


On-Orbit Assembly



ET Debris Shield Concept for MMV Assembly

- Concept---"Raft" expended NSTS External Tanks on-orbit to form debris shield
  - Requires two rows of 6 ET's each
  - Needs power, guidance, attitude control, reboost subsystems
  - Tilt shield into velocity vector to reduce drag
  - Provides work platform for assembly tasks
  - Assembled by NSTS

- Alternatives
  - Bring up debris shield with MMV---150K lbs for SSF-equivalent shield
  - Use MMV aerobrakes as shields---risk flight hardware

ADVANCED CIVIL SPACE SYSTEMS

ET Debris Shield Concept for MMV Assembly (Cont)

BOEING

- Advantages
  - One-third less weight penalty to orbit than separate shield---50K vs 150K lbs
  - Provides greater protection than SSF shield design
  - Debris shield can be completed years before MMV assembly start
  - Provides experience with on-orbit assembly of large structures

- Disadvantages
  - 3-4000 lbs penalty for each STS flight to orbit ET---48K lbs for 12 tanks
    - 1000 lbs for vent, tumble valve, range safety system modifications
    - 2-3000 lbs for OMS propellant
  - Additional 2000 lbs allowed for connecting structure/fill shielding for gaps
  - Debris shield is not continuous
  - High orbital drag (can be flown tilted into velocity vector to minimize drag)
  - May require on-orbit containment of SOFI
    - SOFI may become ablated, charred during ascent
    - UV degradation on-orbit
    - Outgassing
  - Requires development of power, guidance, attitude control, reboost systems

ET Debris Shield Assembly Sequence

- First Orbiter/ET flight
  - First Orbiter/ET taken to MMV assembly orbit
  - Orbiter separates from ET, turns PLB toward ET
  - EVA or RMS attachment of EPS, GNC, RCS, C&T packages
- Subsequent Orbiter/ET flights
  - Orbiter/ET rendezvous with debris shield
  - Attach connecting structure to existing ET hardpoints
  - Orbiter separates from ET, attach ET to debris shield
  - Upgrade/relocate subsystems as debris shield buildup continues
  - Complete initial configuration of 6 ET's/final of 12 ET's
- First MMV assembly flight
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Upgrade subsystems as required for aging, damage
  - Install PRMS
  - Stow MMV components on assembly platform
  - Initiate MMV assembly
- Subsequent MMV assembly flights
  - HLLV/OMV rendezvous with ET debris shield/assembly platform
  - Stow MMV components
  - Continue MMV assembly
  - Upgrade/relocate subsystems as MMV buildup continues
  - Resupply consumables

MMV Manifesting

- Vehicle: HLLV (2 or 3 Stage)
  - Abilities - 2 Stage
    - 10M x 30M Payload Envelope
    - 84 ton capacity
  - Abilities - 3 Stage
    - 7.6M x 30M Payload Envelope (less 3rd Stage)
    - 120 ton capacity
- HLLV Mission One (2 Stage)
  - MTV Habitat Module
  - Mars Surface Payload
  - Assembly Platform Support Equipment
- HLLV Mission Two (2 Stage)
  - MEV Aerobrake Sections
  - MTV Habitat Module Refurbishment/Consumables
- HLLV Mission Three (2 Stage)
  - MEV Aerobrake Sections
  - Assembly Platform Support Equipment

MMV Manifesting (cont'd)

- HLLV Mission Four (2 Stage)
  - MEV Lander Structure
  - Lander Legs
  - Descent System
  - Ascent System
  - Science Payload
  - Airlock
  - Stairs

- HLLV Mission Five (2 Stage)
  - MTV Aerobrake Sections
  - MTV Habitat Module Consumables

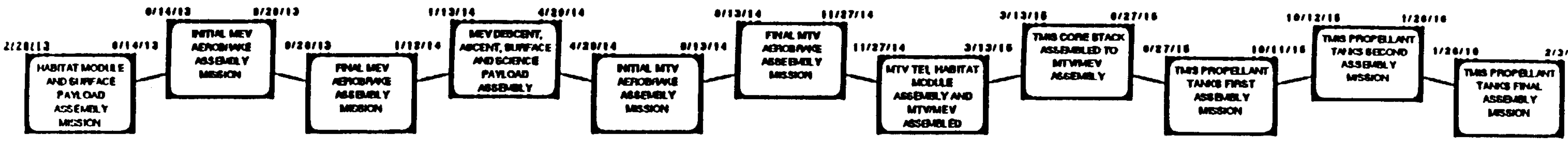
- HLLV Mission Six (2 Stage)
  - MTV Aerobrake Sections
  - Assembly Platform Support Equipment

- HLLV Mission Seven (2 Stage)
  - MTV Trans Earth Injection System
  - MTV Habitat Consumables
  - Assembly Platform Support Equipment

- HLLV Mission Eight (3 Stage)
  - TMI Propellant with Engines

- HLLV Mission Nine thru Eleven (3 Stage)
  - TMI Propellant

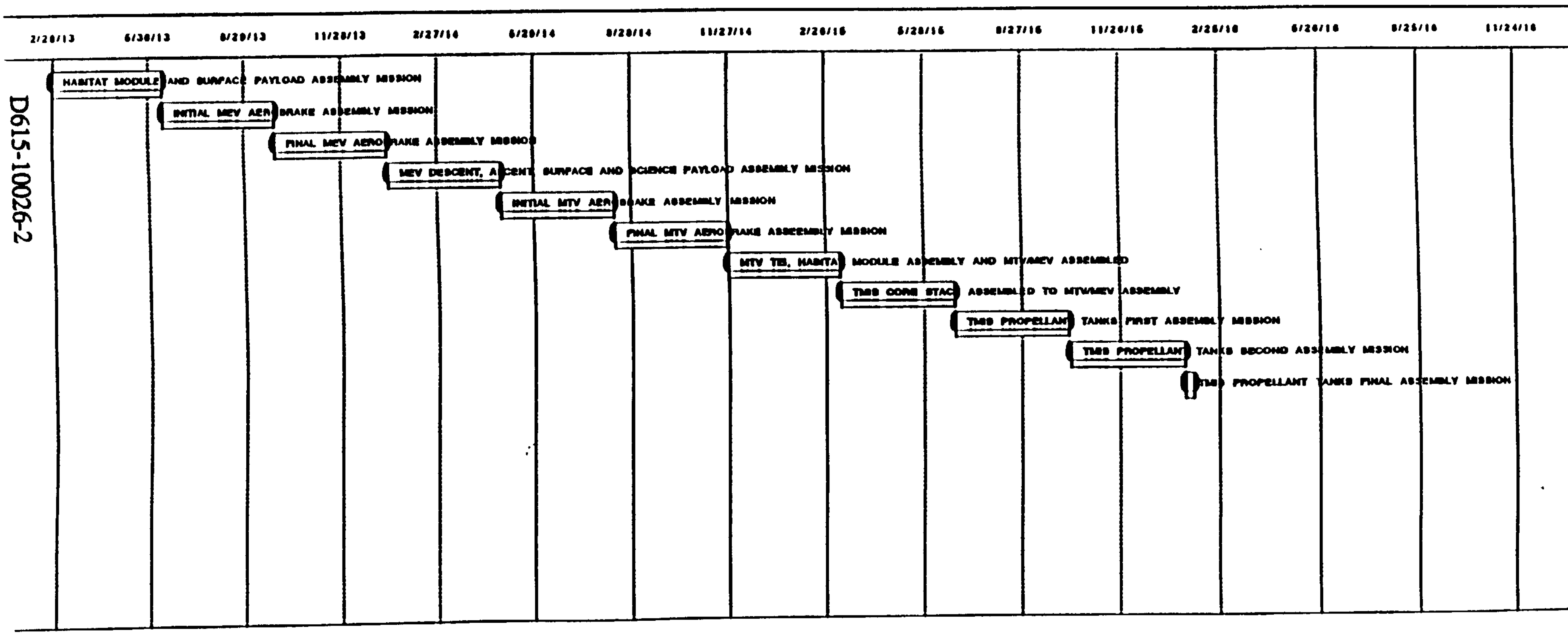
MMV TOP ASSEMBLY



- Smallest unit of time is 1 man-hour
- 16 man-hours = 1 man-day of Assembly Duration

BASELINE DURATIONS:
- HLLV Launch = .5 man-day
- HLLV achieves stable orbit = .25 man-day
- OMV deploys from/to Freedom = .5 man-day
- OMV berths to components = .25 man-day
- Unstow and power up Robotics = .06 man-day
- Robotic verification = .12 man-day
- HLLV deploys components = .06 man-day
- OMV transfers components = .25 man-day
- Robotic tasks = .06 man-day
- EVA/Robotic Contingency = .5 man-day
- Component Inspection = .12 man-day
- Component Test = .25 man-day
- Subassemblies to stand-by mode = .5 man-day
- Mechanical Fastening of components = .18 man-day

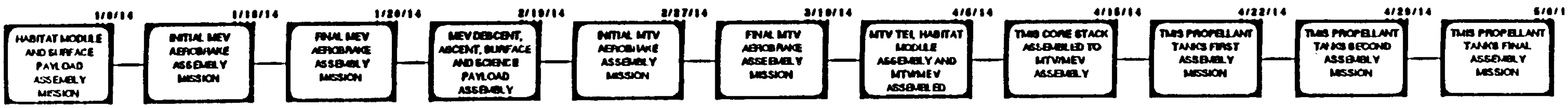
On-Orbit Assembly



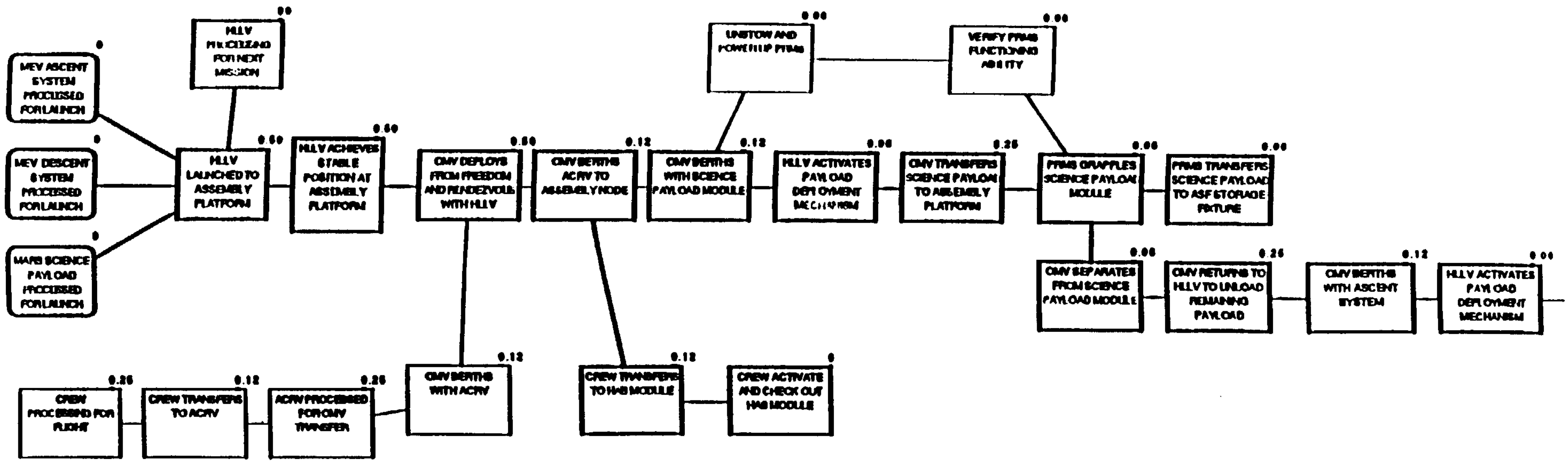
On-Orbit Assembly

Name	Earliest Start	Earliest Finish	Subproject	Days
HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION	2/28/13	6/14/13	HLLV MISSION ONE	106
INITIAL MEV AEROBRAKE ASSEMBLY MISSION	6/14/13	9/28/13	HLLV MISSION TWO	106
FINAL MEV AEROBRAKE ASSEMBLY MISSION	9/28/13	1/12/14	HLLV MISSION THREE	106
MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY	1/13/14	4/29/14	HLLV MISSION FOUR	106
INITIAL MTV AEROBRAKE ASSEMBLY MISSION	4/29/14	8/13/14	HLLV MISSION FIVE	106
FINAL MTV AEROBRAKE ASSEEMBLY MISSION	8/13/14	11/27/14	HLLV MISSION SIX	106
MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED	11/27/14	3/13/15	HLLV MISSION SEVEN	106
TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY	3/13/15	6/27/15	HLLV MISSION EIGHT	106
TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION	6/27/15	10/11/15	HLLV MISSION NINE	106
TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION	10/12/15	1/26/16	HLLV MISSION TEN	106
TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION	1/26/16	2/3/16	HLLV MISSION ELEVEN	8

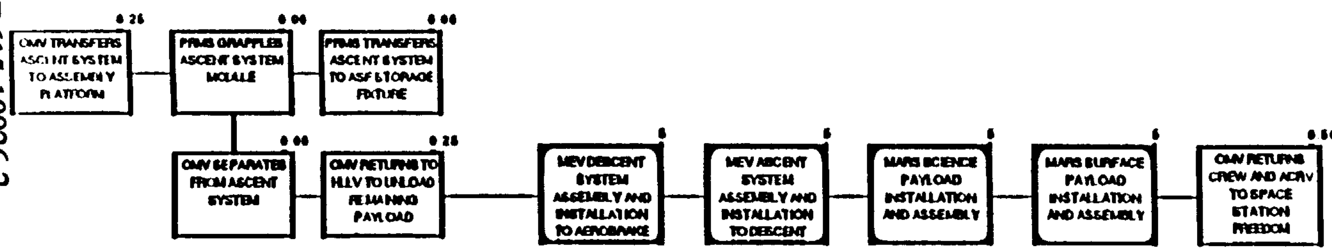
On-Orbit Assembly



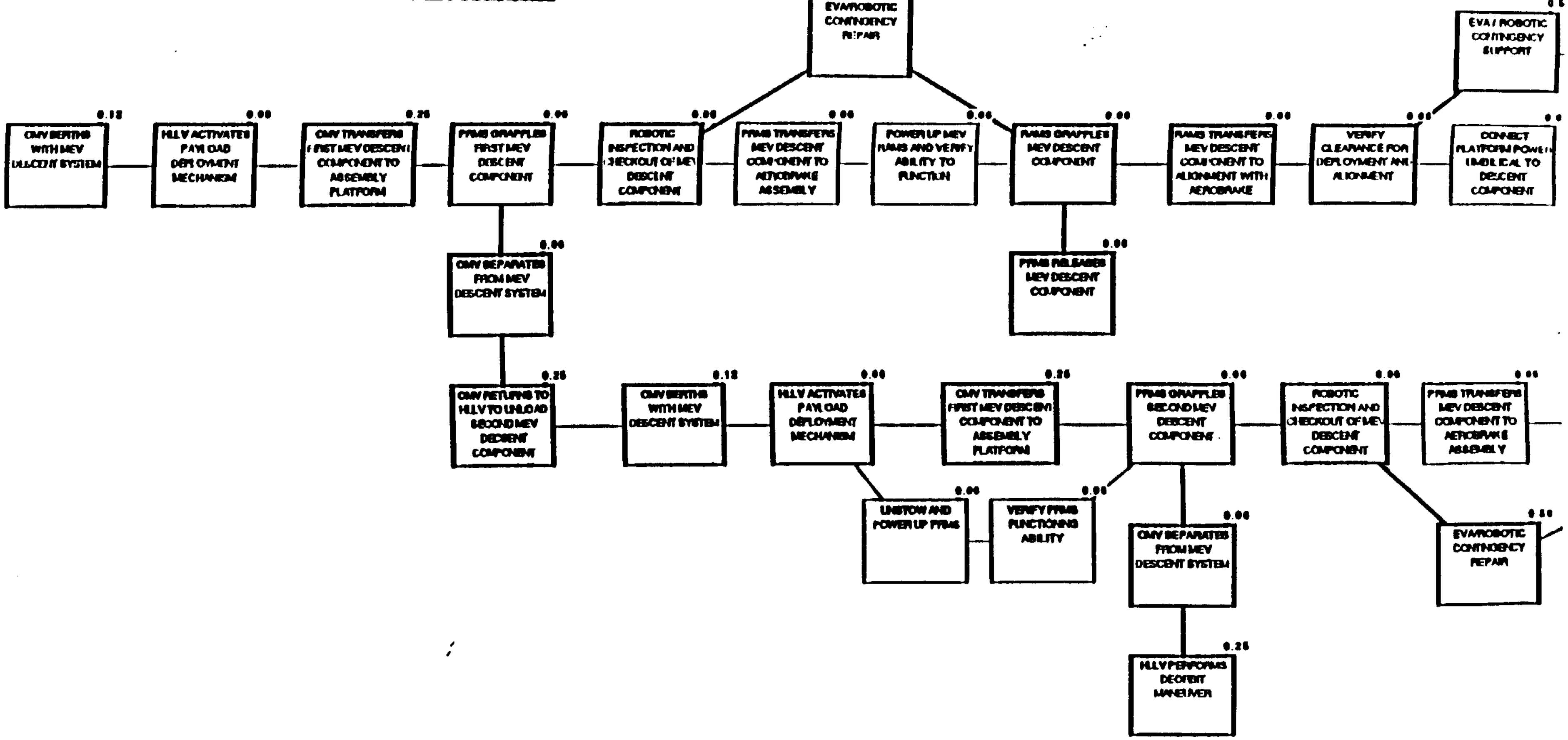
HLLV MISSION FOUR



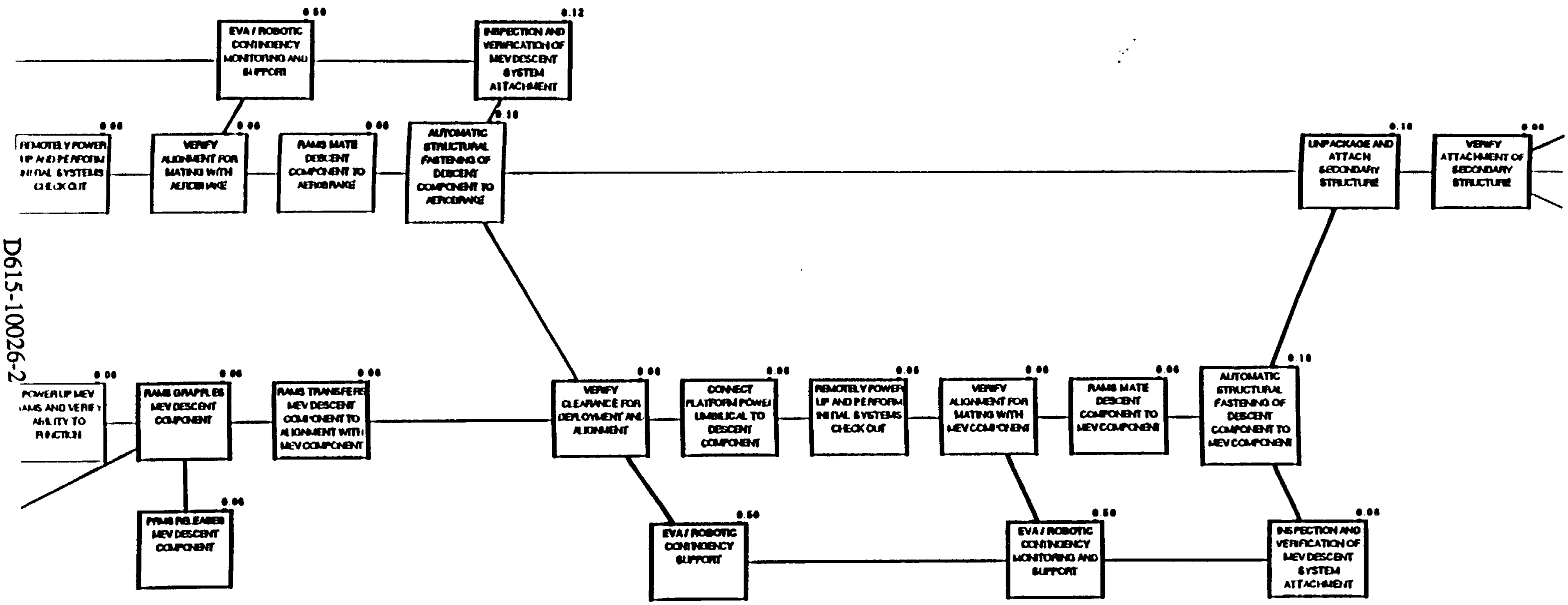
On-Orbit Assembly



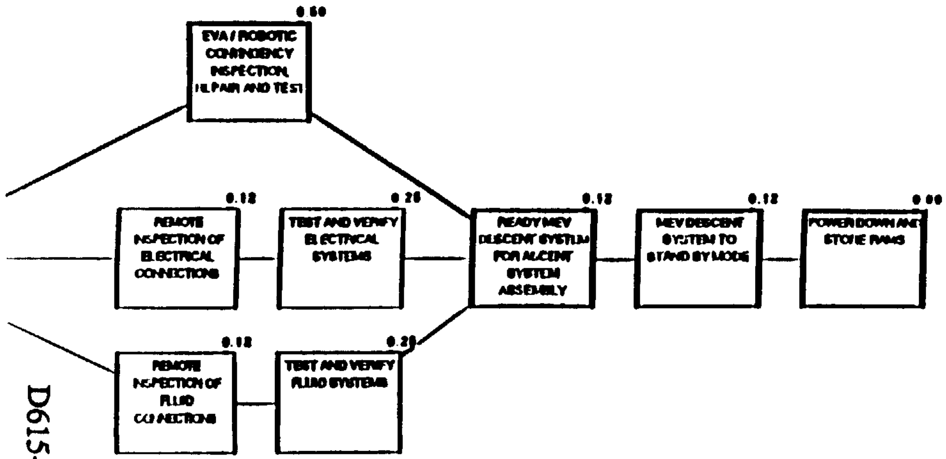
MEV DESCENT SYSTEM



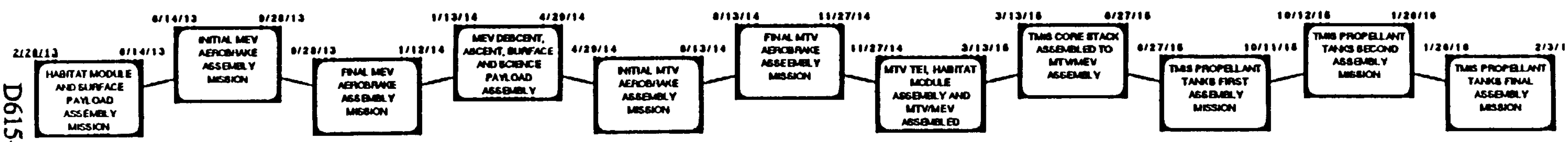
On-Orbit Assembly



On-Orbit Assembly



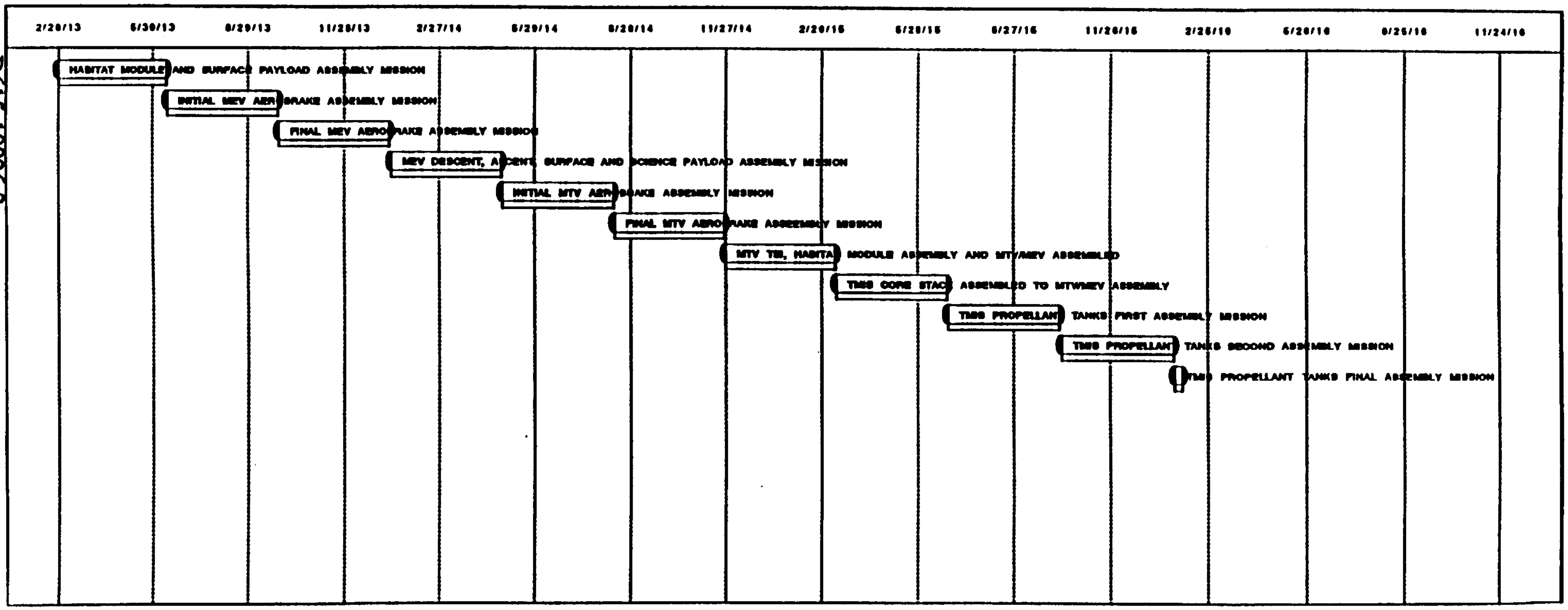
MMV TOP ASSEMBLY



- Smallest unit of time is 1 man-hour
- 16 man-hours = 1 man-day of Assembly Duration

BASELINE DURATIONS:
- HLLV Launch = .5 man-day
- HLLV achieves stable orbit = .25 man-day
- OMV deploys from/to Freedom = .5 man-day
- OMV berths to components = .25 man-day
- Unstow and power up Robotics = .06 man-day
- Robotic verification = .12 man-day
- HLLV deploys components = .06 man-day
- OMV transfers components = .25 man-day
- Robotic tasks = .06 man-day
- EVA/Robotic Contingency = .5 man-day
- Component Inspection = .12 man-day
- Component Test = .25 man-day
- Subassemblies to stand-by mode = .5 man-day
- Mechanical Fastening of components = .18 man-day

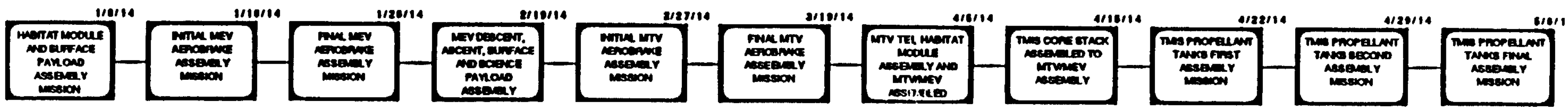
On-Orbit Assembly



On-Orbit Assembly

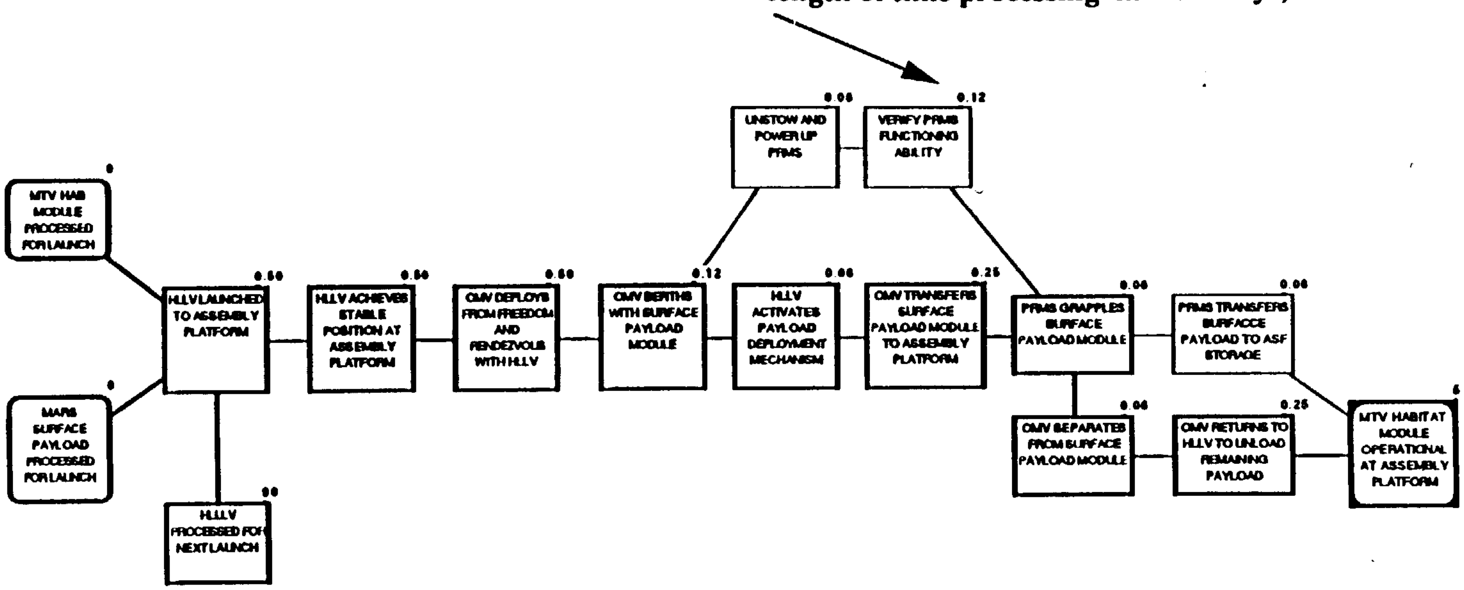
Name	Earliest Start	Earliest Finish	Subproject	Days
HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION	2/28/13	6/14/13	HLLV MISSION ONE	106
INITIAL MEV AEROBRAKE ASSEMBLY MISSION	6/14/13	9/28/13	HLLV MISSION TWO	106
FINAL MEV AEROBRAKE ASSEMBLY MISSION	9/28/13	1/12/14	HLLV MISSION THREE	106
MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY	1/13/14	4/29/14	HLLV MISSION FOUR	106
INITIAL MTV AEROBRAKE ASSEMBLY MISSION	4/29/14	8/13/14	HLLV MISSION FIVE	106
FINAL MTV AEROBRAKE ASSEEMBLY MISSION	8/13/14	11/27/14	HLLV MISSION SIX	106
MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED	11/27/14	3/13/15	HLLV MISSION SEVEN	106
TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY	3/13/15	6/27/15	HLLV MISSION EIGHT	106
TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION	6/27/15	10/11/15	HLLV MISSION NINE	106
TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION	10/12/15	1/26/16	HLLV MISSION TEN	106
TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION	1/26/16	2/3/16	HLLV MISSION ELEVEN	8

On-Orbit Assembly

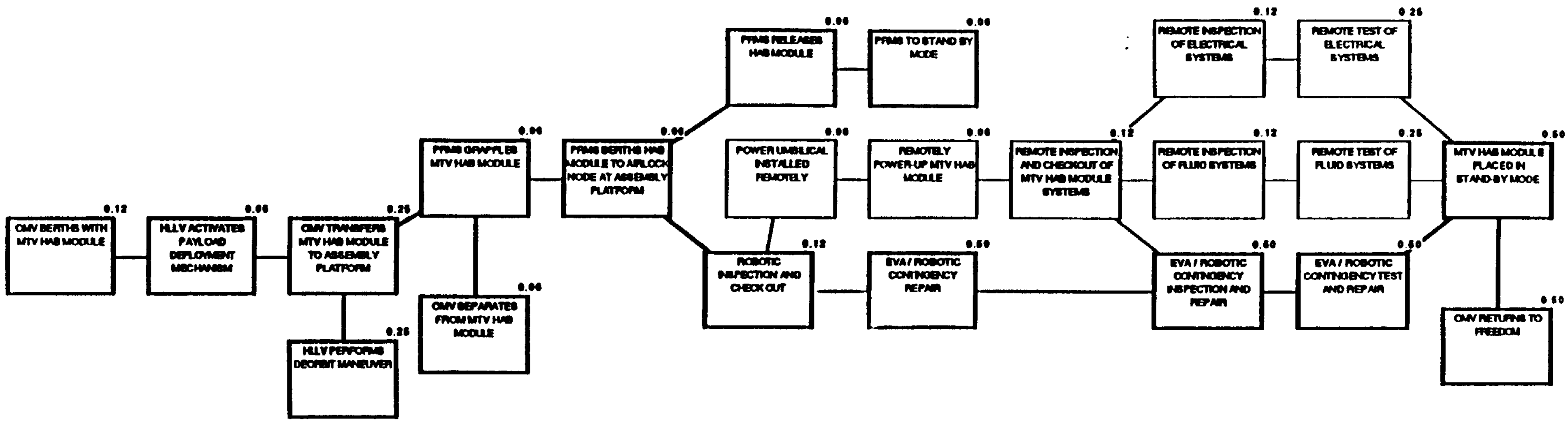


HLLV MISSION ONE

length of time processing in workdays, based on a 16 hour workday

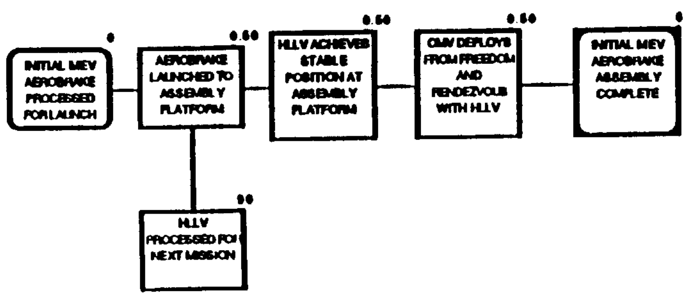


MTV HABITAT MODULE

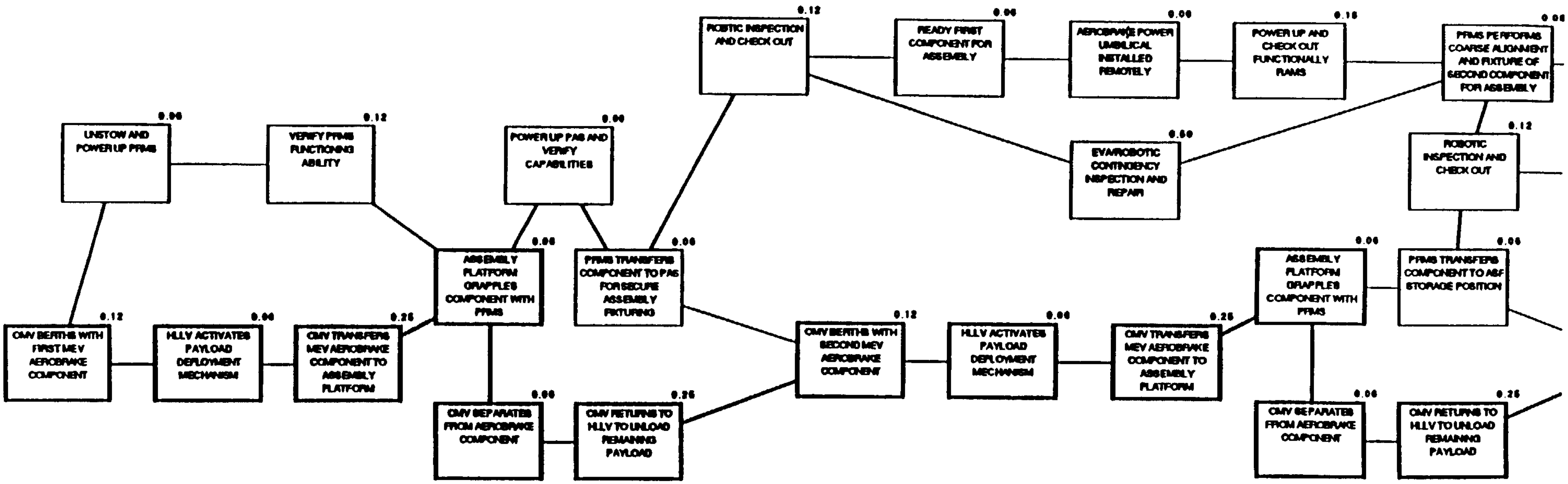


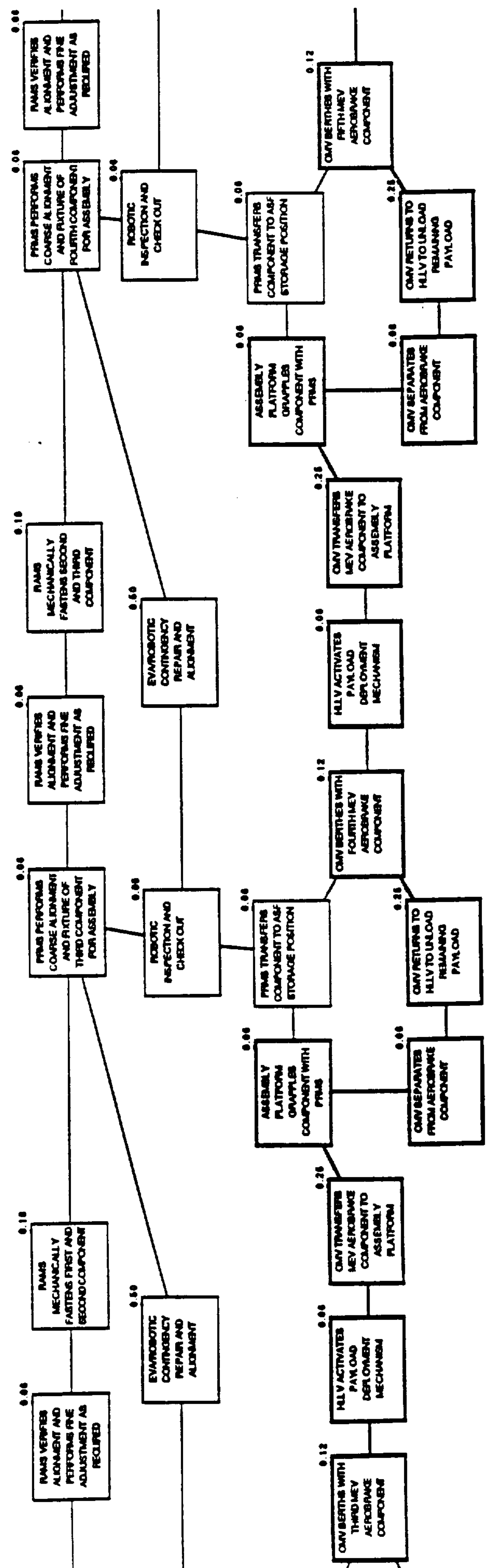
6

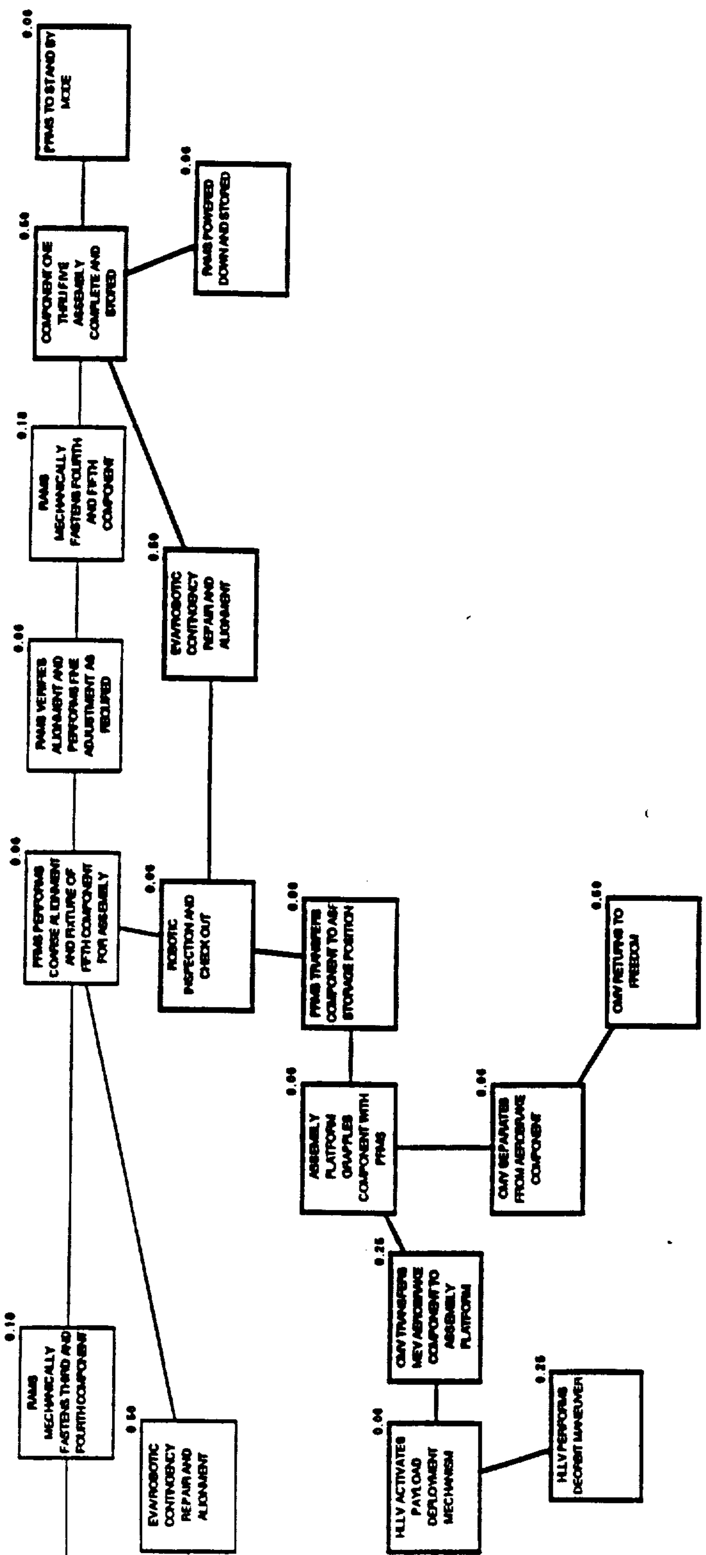
HLLV MISSION TWO



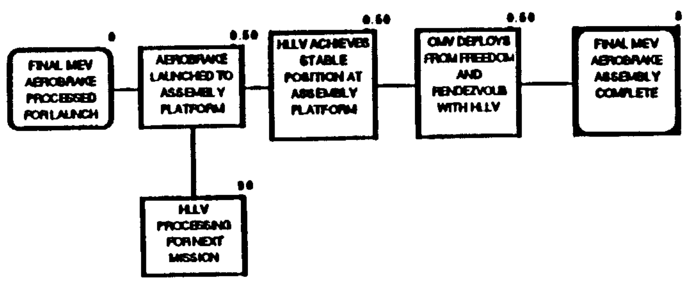
MEV AEROBRAKE MISSION ONE



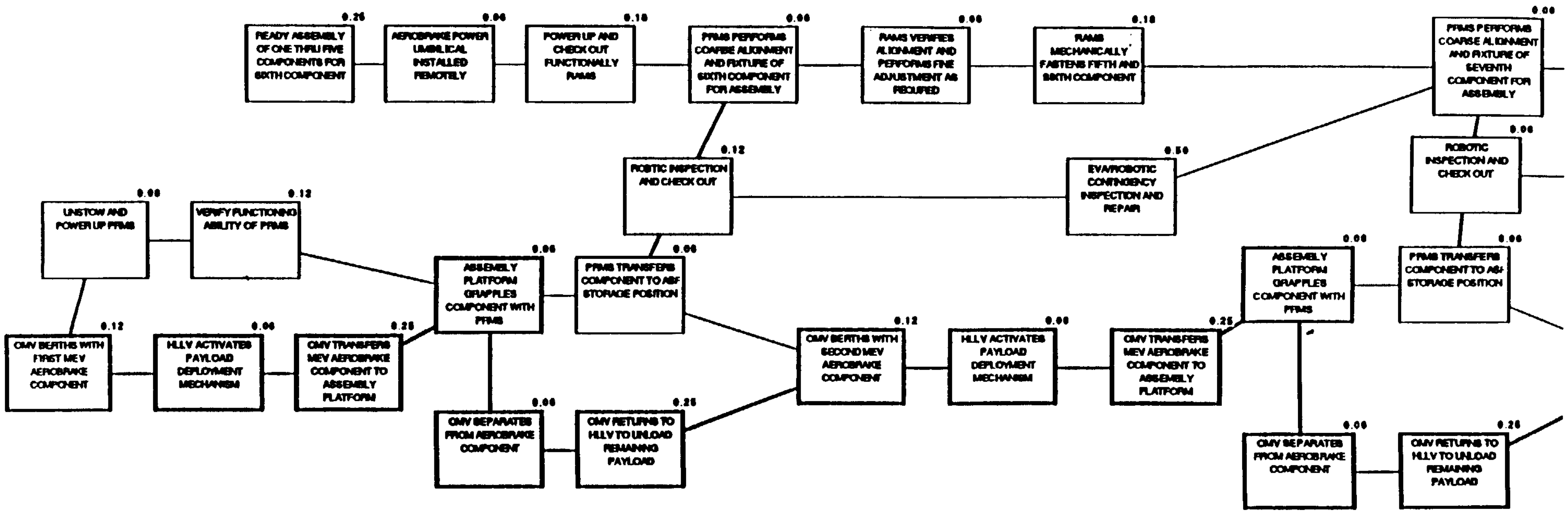

RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
RMS MECHANICALLY FASTENS FIRST AND SECOND COMPONENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF THIRD COMPONENT FOR ASSEMBLY
RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
RMS MECHANICALLY FASTENS SECOND AND THIRD COMPONENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FOURTH COMPONENT FOR ASSEMBLY
RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
ROBOTIC INSPECTION AND CHECK OUT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
ROBOTIC INSPECTION AND CHECK OUT
OMV BERTHS WITH THIRD MEV AEROBRAKE COMPONENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
PRMS TRANSFERS COMPONENT TO AS/F STORAGE POSITION
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
OMV BERTHS WITH FOURTH MEV AEROBRAKE COMPONENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
PRMS TRANSFERS COMPONENT TO AS/F STORAGE POSITION
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
OMV BERTHS WITH FIFTH MEV AEROBRAKE COMPONENT

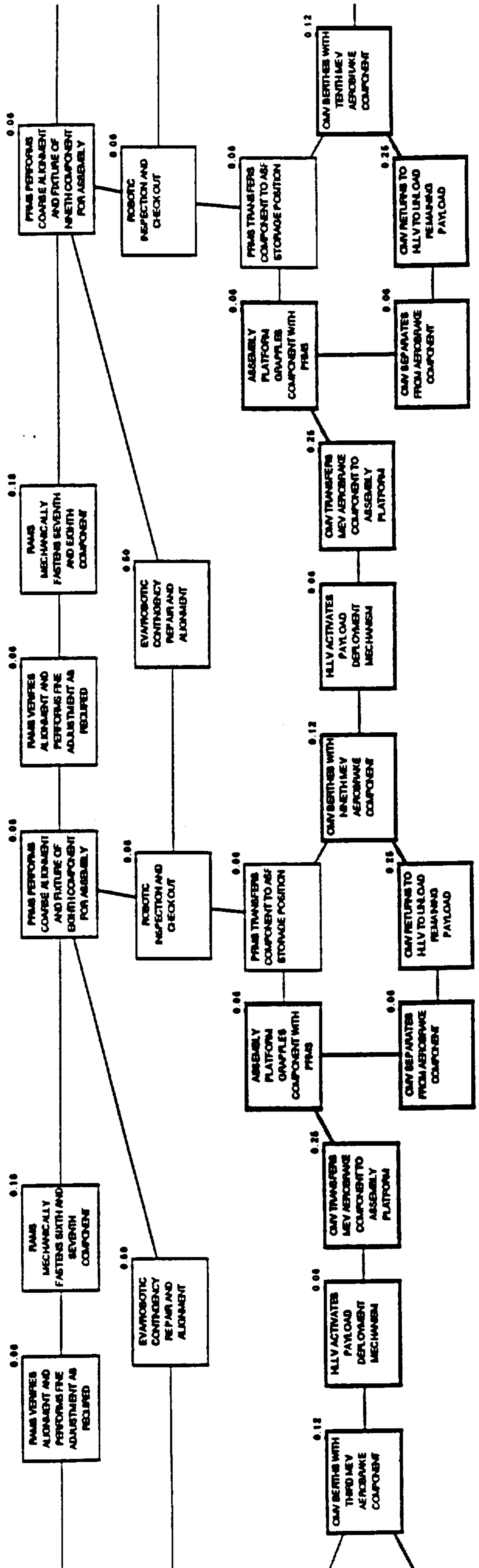

PFMS TO STAND BY MODE
COMPONENT ONE THRU FIVE ASSEMBLY COMPLETE AND STORED
PFMS POWERED DOWN AND STORED
PFMS MECHANICALLY FASTENS FOURTH AND FIFTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
PFMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
PFMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FIFTH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECK OUT
PFMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
OMV RETURNS TO FREEDOM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PFMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
OMV TRANSFERS NEW AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
PFMS MECHANICALLY FASTENS THIRD AND FOURTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
HLLV PERFORMS DEORBIT MANEUVER

HLLV MISSION THREE

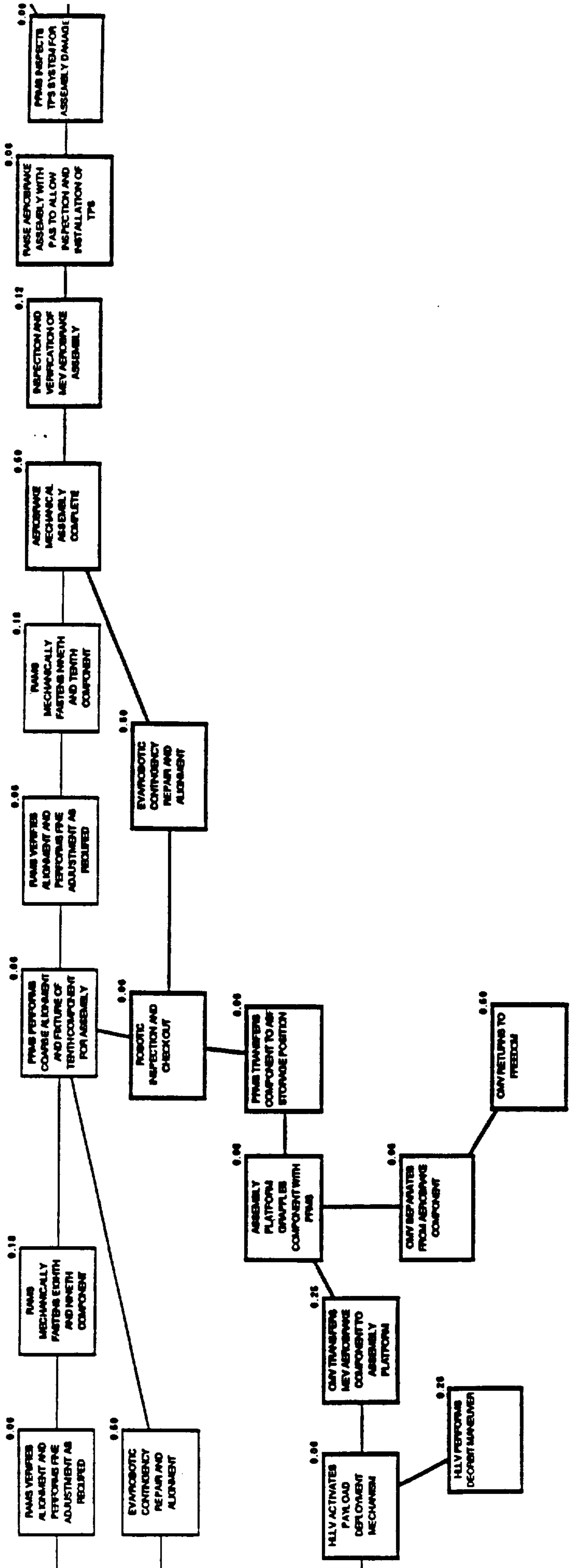


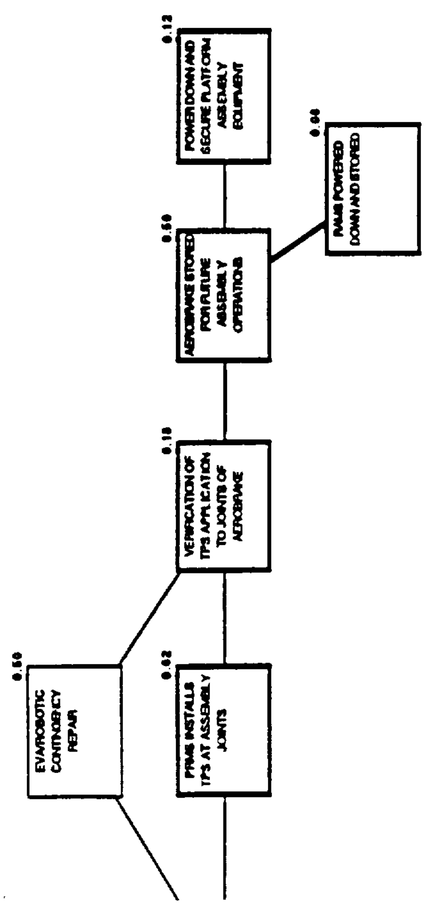
MEV AEROBRAKE MISSION TWO



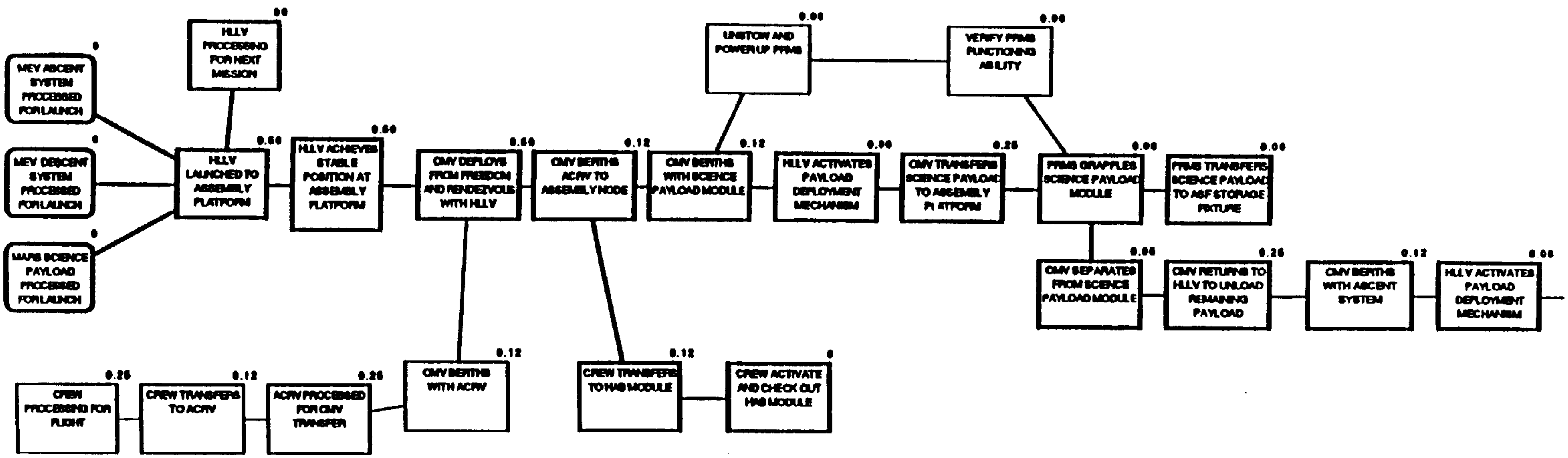


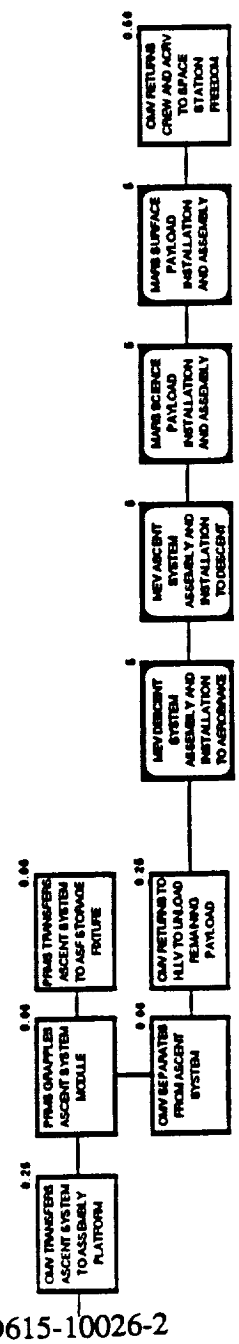
PRMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
PRMS MECHANICALLY FASTENS SIXTH AND SEVENTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
OMV BERTHS WITH THIRD MEV AEROBRAKE COMPONENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF EIGHTH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECKOUT
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
OMV BERTHS WITH NINETH MEV AEROBRAKE COMPONENT
PRMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
PRMS MECHANICALLY FASTENS SEVENTH AND EIGHTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF NINETH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECKOUT
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
OMV BERTHS WITH TENTH MEV AEROBRAKE COMPONENT


PRMS INSPECTS TPS SYSTEM FOR ASSEMBLY DAMAGE
RAISE AEROBRAKE ASSEMBLY WITH PAS TO ALLOW INSPECTION AND INSTALLATION OF TPS
0.12
INSPECTION AND VERIFICATION OF MEV AEROBRAKE ASSEMBLY
0.50
AEROBRAKE MECHANICAL ASSEMBLY COMPLETE
0.10
PRMS MECHANICALLY FASTENS NINETH AND TENTH COMPONENT
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
PRMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF TENTH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECK OUT
PRMS TRANSFERS COMPONENT TO AIR STORAGE POSITION
0.50
OMV RETURNS TO FREEDOM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.10
PRMS MECHANICALLY FASTENS EIGHTH AND NINETH COMPONENT
0.25
OMV TRANSFERS MEV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
HLLV PERFORMS DEORBIT MANEUVER
PRMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.50
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM


0.50
EVA/ROBOTIC CONTINGENCY REPAIR
0.02
PRMS INSTALLS TPS AT ASSEMBLY JOINTS
VERIFICATION OF TPS APPLICATION TO JOINTS OF AEROBRAKE
0.50
AEROBRAKE STORED FOR FUTURE ASSEMBLY OPERATIONS
0.12
POWER DOWN AND SECURE PLATFORM ASSEMBLY EQUIPMENT
PRMS POWERED DOWN AND STORED

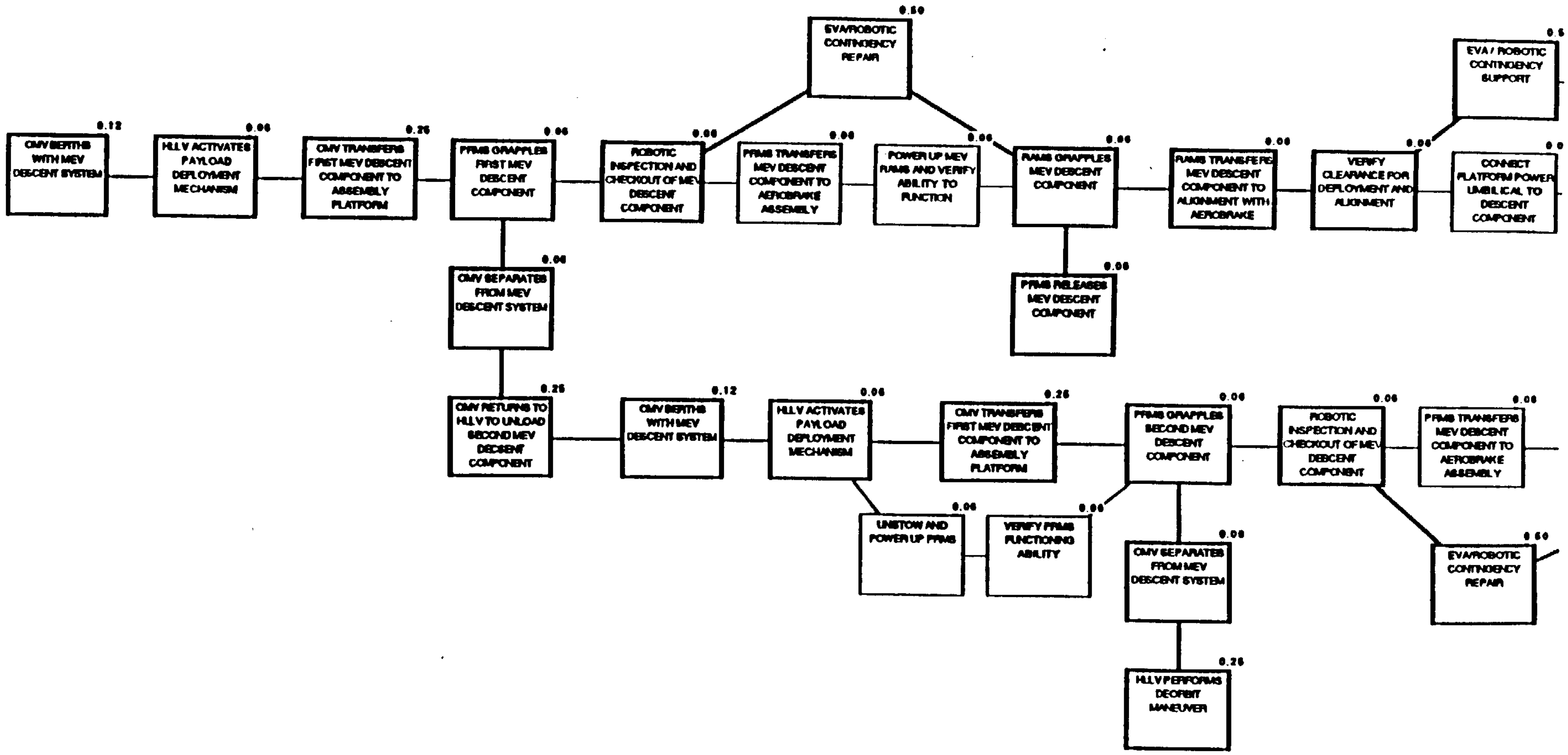
HLLV MISSION FOUR

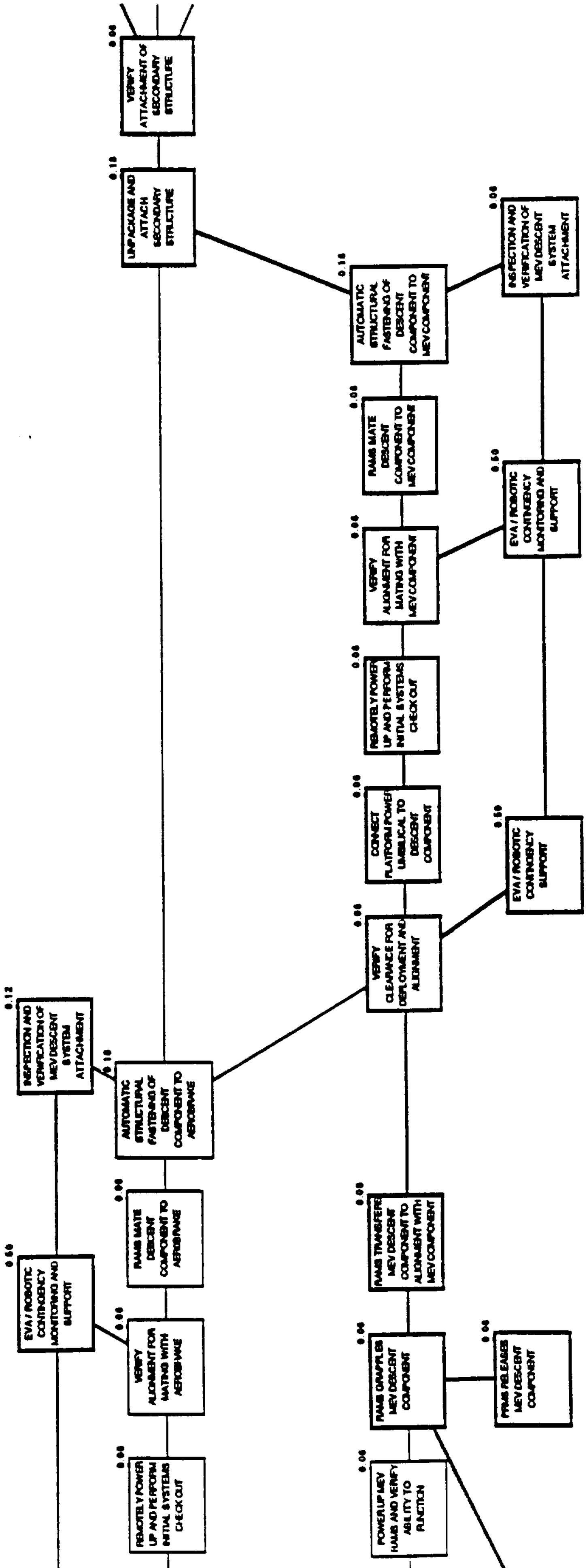


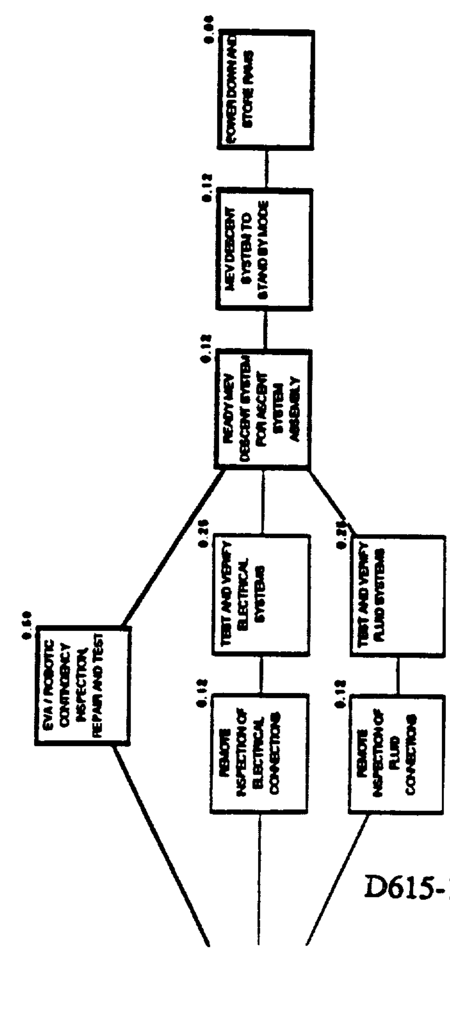


0.25
OMV TRANSFERS ASCENT SYSTEM TO ASSEMBLY PLATFORM
0.06
PRMS GRAPPLES ASCENT SYSTEM MODULE
0.06
PRMS TRANSFERS ASCENT SYSTEM TO ASF STORAGE FIXTURE
0.06
OMV SEPARATES FROM ASCENT SYSTEM
0.25
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
5
MEV DESCENT SYSTEM ASSEMBLY AND INSTALLATION TO AEROBRAKE
5
MEV ASCENT SYSTEM ASSEMBLY AND INSTALLATION TO DESCENT
5
MARS SCIENCE PAYLOAD INSTALLATION AND ASSEMBLY
5
MARS SURFACE PAYLOAD INSTALLATION AND ASSEMBLY
0.50
OMV RETURNS CREW AND ACRV TO SPACE STATION FREEDOM

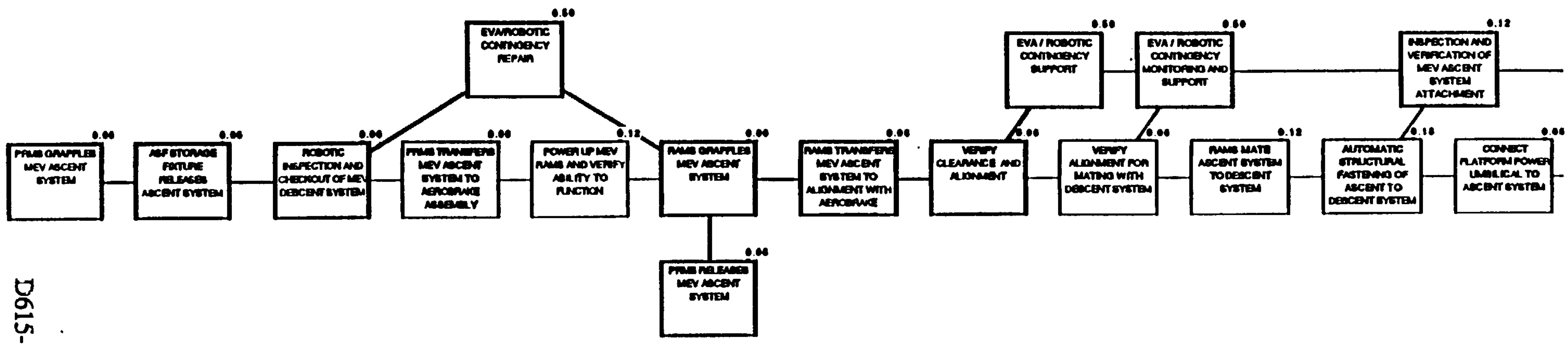
MEV DESCENT SYSTEM

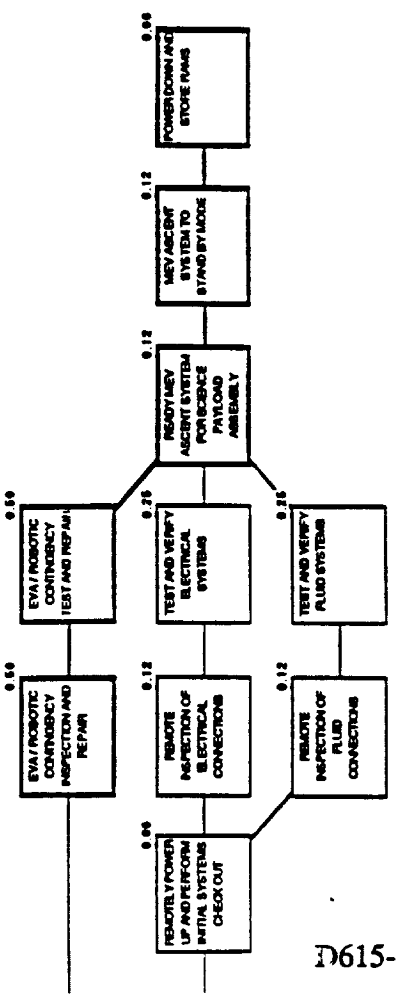



REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
VERIFY ALIGNMENT FOR MATING WITH AEROBRAKE
RAMS MATE DESCENT COMPONENT TO AEROBRAKE
AUTOMATIC STRUCTURAL FASTENING OF DESCENT COMPONENT TO AEROBRAKE
EVA / ROBOTIC CONTINGENCY MONITORING AND SUPPORT
INSPECTION AND VERIFICATION OF MEV DESCENT SYSTEM ATTACHMENT
UNPACKAGE AND ATTACH SECONDARY STRUCTURE
VERIFY ATTACHMENT OF SECONDARY STRUCTURE
POWER UP MEV RAMS AND VERIFY ABILITY TO FUNCTION
RAMS GRAPPLES MEV DESCENT COMPONENT
RMS RELEASES MEV DESCENT COMPONENT
RAMS TRANSFERS MEV DESCENT COMPONENT TO ALIGNMENT WITH MEV COMPONENT
VERIFY CLEARANCE FOR DEPLOYMENT AND ALIGNMENT
EVA / ROBOTIC CONTINGENCY SUPPORT
CONNECT PLATFORM POWER UMBILICAL TO DESCENT COMPONENT
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
VERIFY ALIGNMENT FOR MATING WITH MEV COMPONENT
EVA / ROBOTIC CONTINGENCY MONITORING AND SUPPORT
RAMS MATE DESCENT COMPONENT TO MEV COMPONENT
AUTOMATIC STRUCTURAL FASTENING OF DESCENT COMPONENT TO MEV COMPONENT
INSPECTION AND VERIFICATION OF MEV DESCENT SYSTEM ATTACHMENT



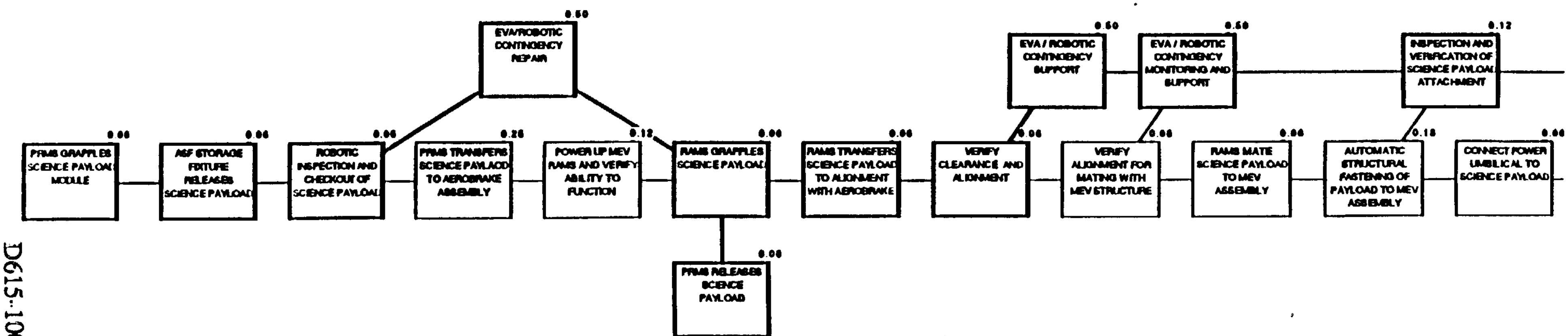
MEV ASCENT SYSTEM

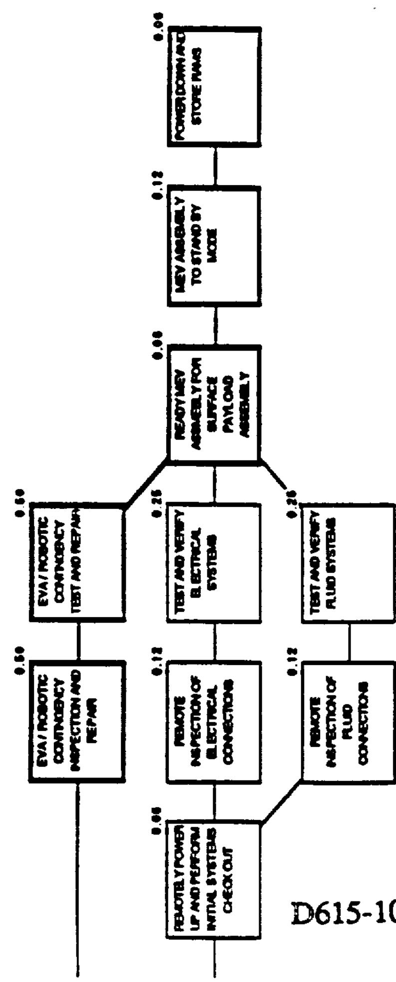




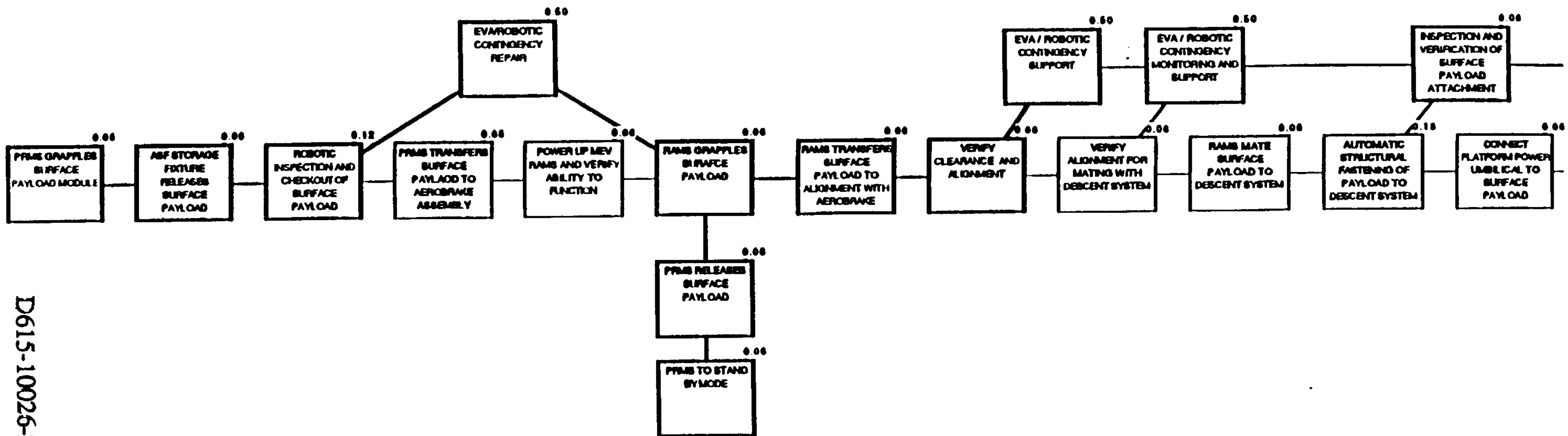
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
REMOTE INSPECTION OF FLUID CONNECTIONS
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
TEST AND VERIFY ELECTRICAL SYSTEMS
TEST AND VERIFY FLUID SYSTEMS
READY MEV ASCENT SYSTEM FOR SCIENCE PAYLOAD ASSEMBLY
MEV ASCENT SYSTEM TO STAND BY MODE
POWER DOWN AND STORE RAMS

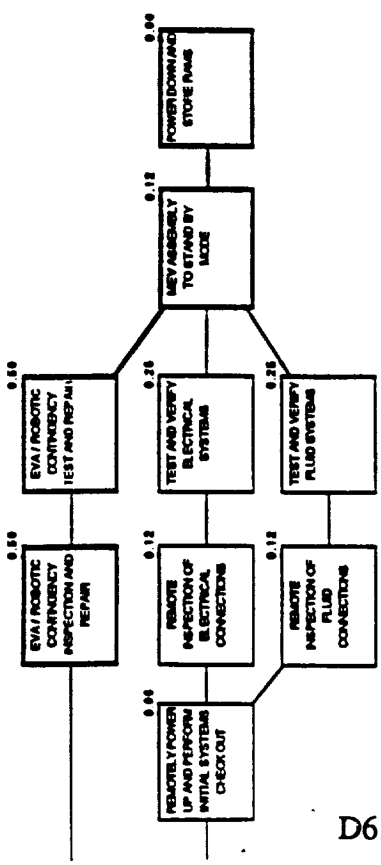
MARS SCIENCE PAYLOAD



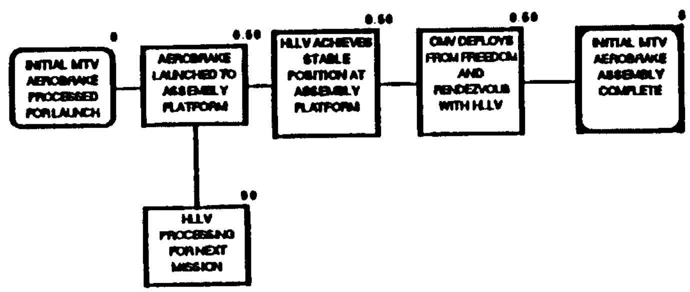

REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
READY MEV ASSEMBLY FOR SURFACE PAYLOAD ASSEMBLY
0.12
MEV ASSEMBLY TO STAND BY MODE
POWER DOWN AND STORE RAMS

MARS SURFACE PAYLOAD

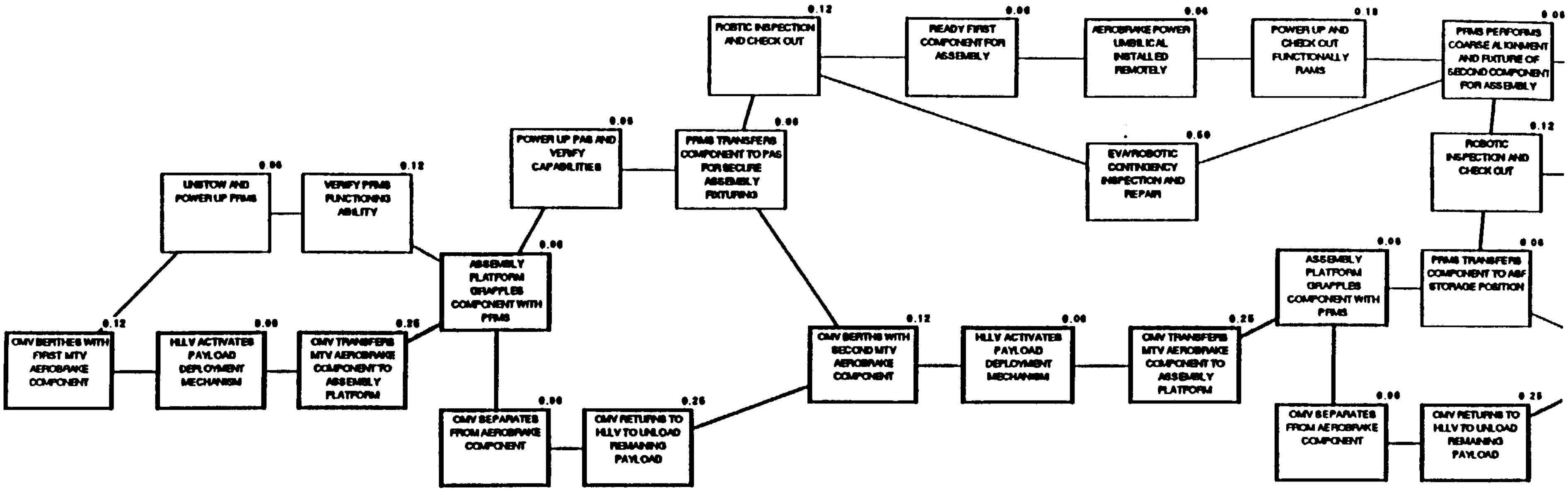


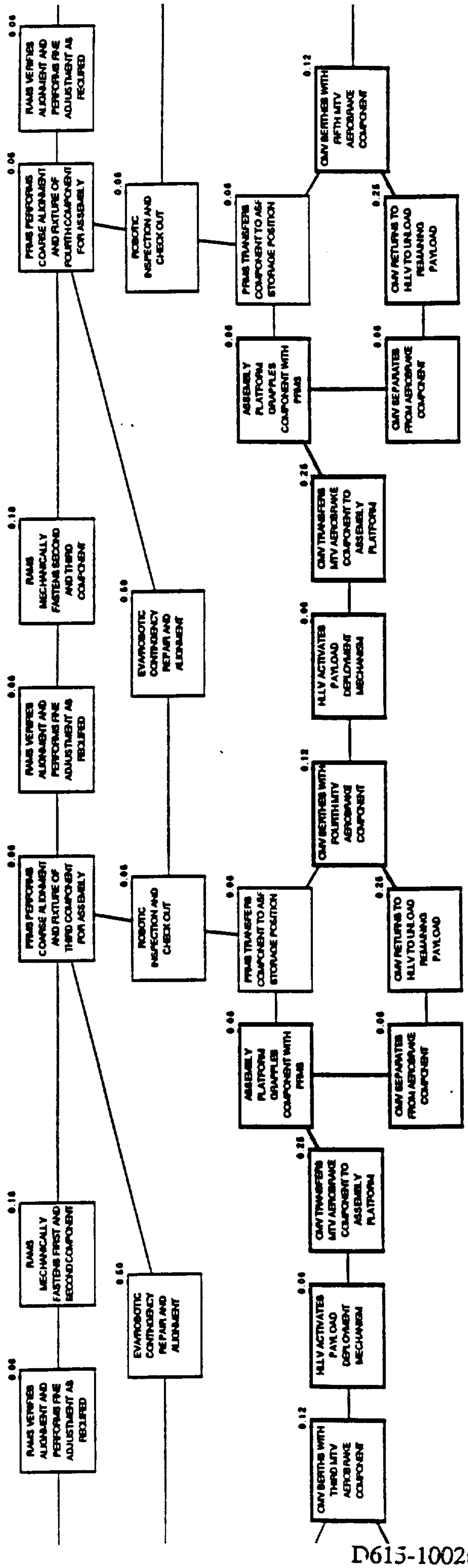

REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
REMOTE INSPECTION OF FLUID CONNECTIONS
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
TEST AND VERIFY ELECTRICAL SYSTEMS
TEST AND VERIFY FLUID SYSTEMS
MEV ASSEMBLY TO STAND BY MODE
POWER DOWN AND STORE RAMS

HLLV MISSION FIVE

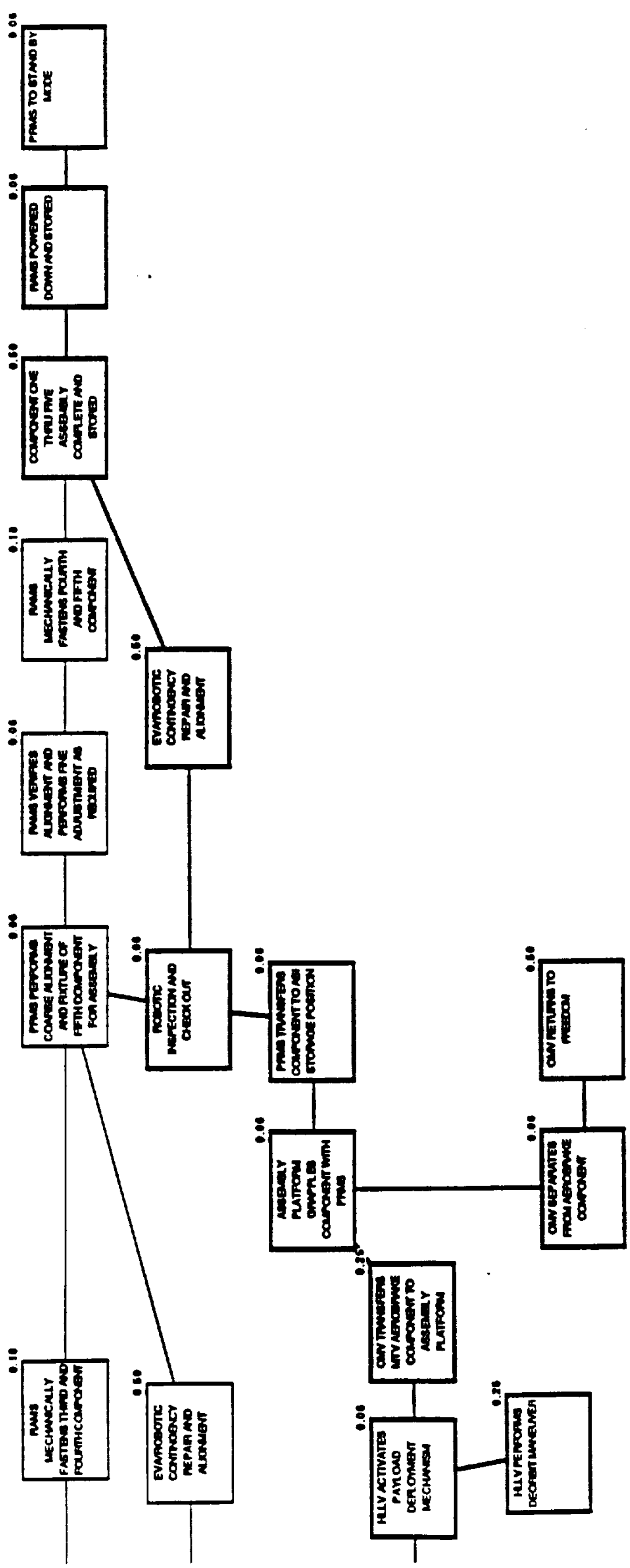


MTV AEROBRAKE MISSION ONE

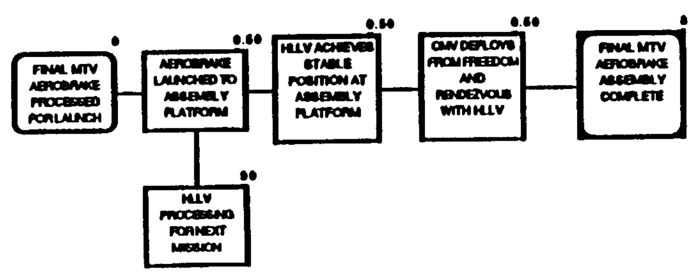




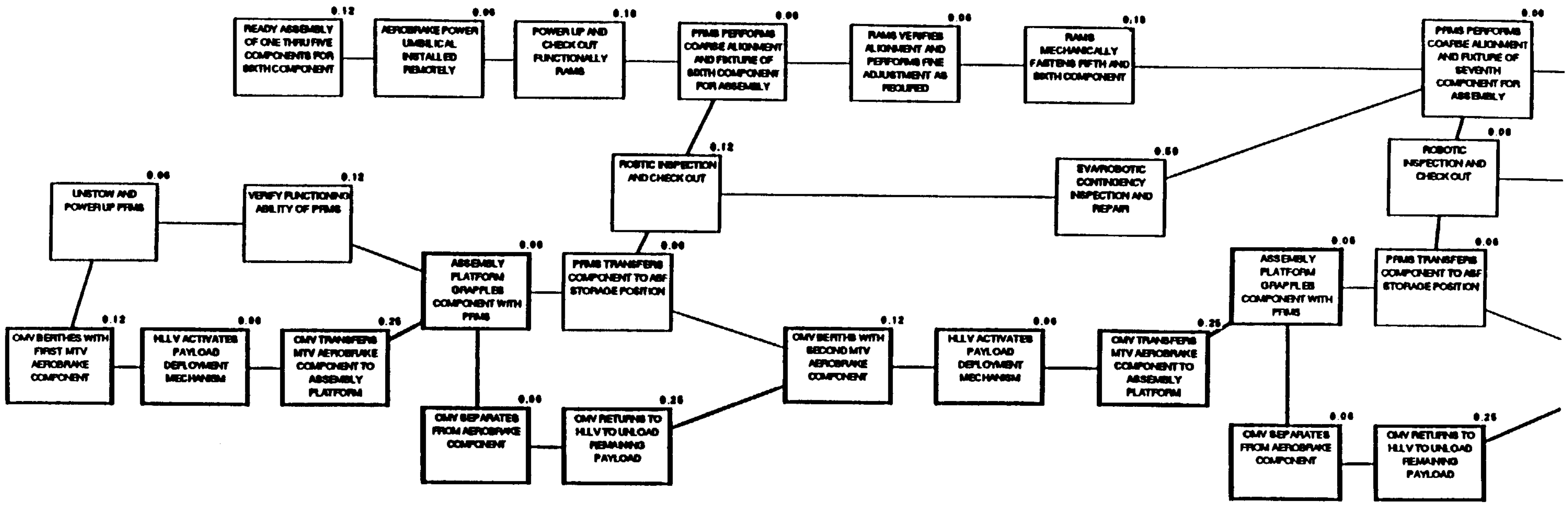
RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RMS MECHANICALLY FASTENS FIRST AND SECOND COMPONENT
0.10
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF THIRD COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
RMS MECHANICALLY FASTENS SECOND AND THIRD COMPONENT
0.10
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FOURTH COMPONENT FOR ASSEMBLY
0.06
ROBOTIC INSPECTION AND CHECK OUT
0.06
RMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
0.06
OMV BERTHS WITH THIRD MTV AEROBRAKE COMPONENT
0.12
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
PRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
0.06
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
OMV BERTHS WITH FOURTH MTV AEROBRAKE COMPONENT
0.12
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
0.06
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
0.06
OMV SEPARATES FROM AEROBRAKE COMPONENT
0.06
PRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
0.06
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
0.25
OMV BERTHS WITH FIFTH MTV AEROBRAKE COMPONENT
0.12

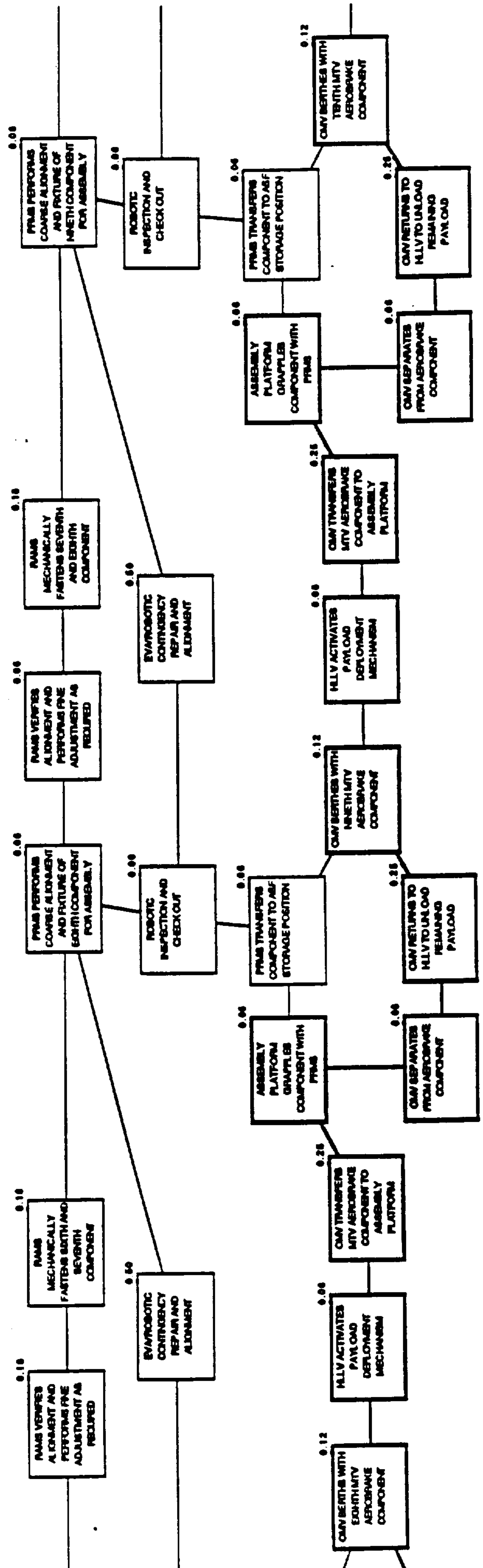

PRMS TO STAND BY MODE
RAMS POWERED DOWN AND STORED
COMPONENT ONE THRU FIVE ASSEMBLY COMPLETE AND STORED
RAMS MECHANICALLY FASTENS FOURTH AND FIFTH COMPONENT
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF FIFTH COMPONENT FOR ASSEMBLY
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
ROBOTIC INSPECTION AND CHECK OUT
PRMS TRANSFERS COMPONENT TO ASI STORAGE POSITION
OMV RETURNS TO FREEDOM
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
RAMS MECHANICALLY FASTENS THIRD AND FOURTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
HLLV PERFORMS DEORBIT MANEUVER

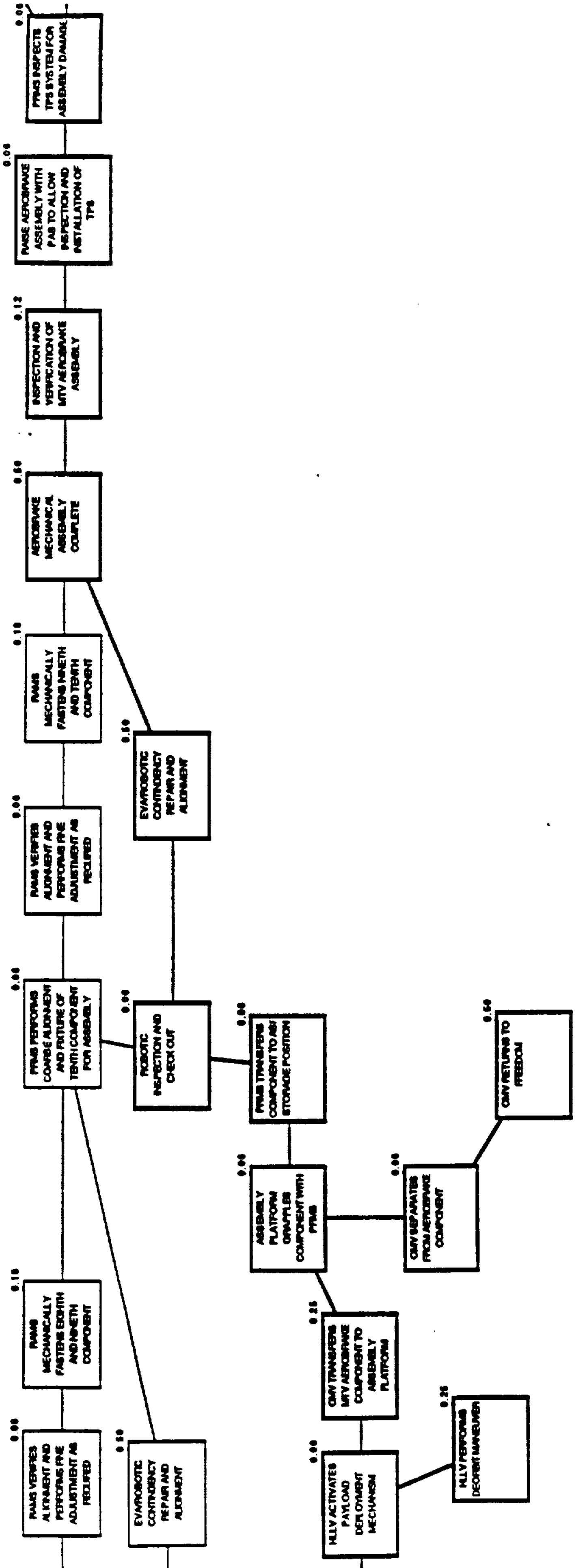
HLLV MISSION SIX

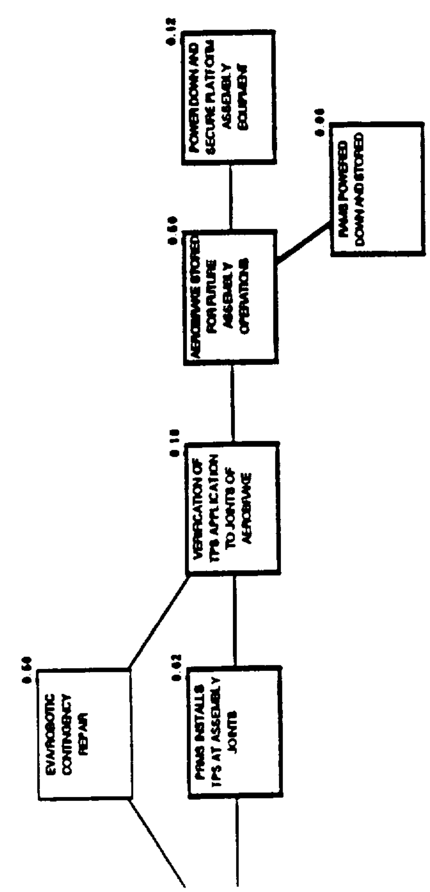


MTV AEROBRAKE MISSION TWO

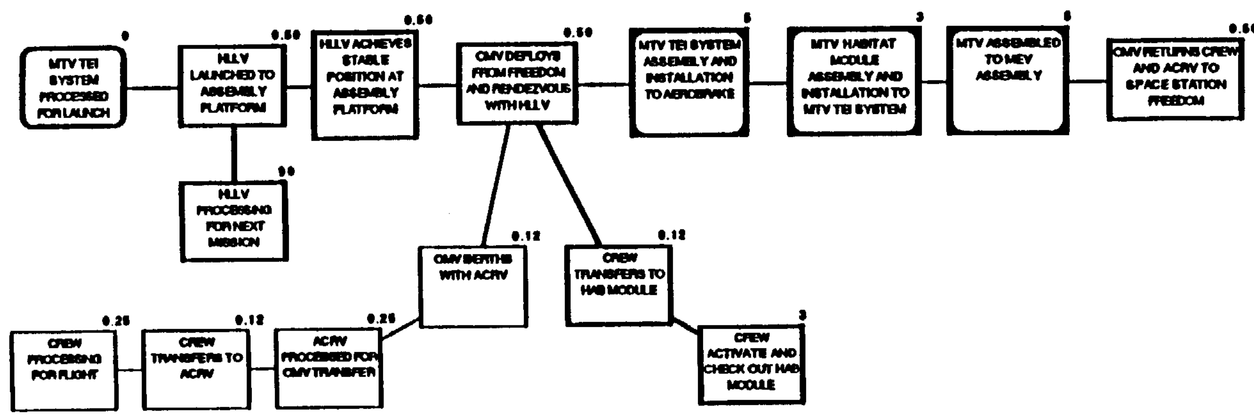



PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF NINETH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECK OUT
PRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
OMV BERTHES WITH TENTH MTV AEROBRAKE COMPONENT
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
RAMS MECHANICALLY FASTENS SEVENTH AND EIGHTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
OMV BERTHES WITH NINETH MTV AEROBRAKE COMPONENT
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF EIGHTH COMPONENT FOR ASSEMBLY
ROBOTIC INSPECTION AND CHECK OUT
PRMS TRANSFERS COMPONENT TO A&F STORAGE POSITION
OMV RETURNS TO HLLV TO UNLOAD REMAINING PAYLOAD
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
OMV SEPARATES FROM AEROBRAKE COMPONENT
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
RAMS MECHANICALLY FASTENS SIXTH AND SEVENTH COMPONENT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
OMV BERTHES WITH EIGHTH MTV AEROBRAKE COMPONENT

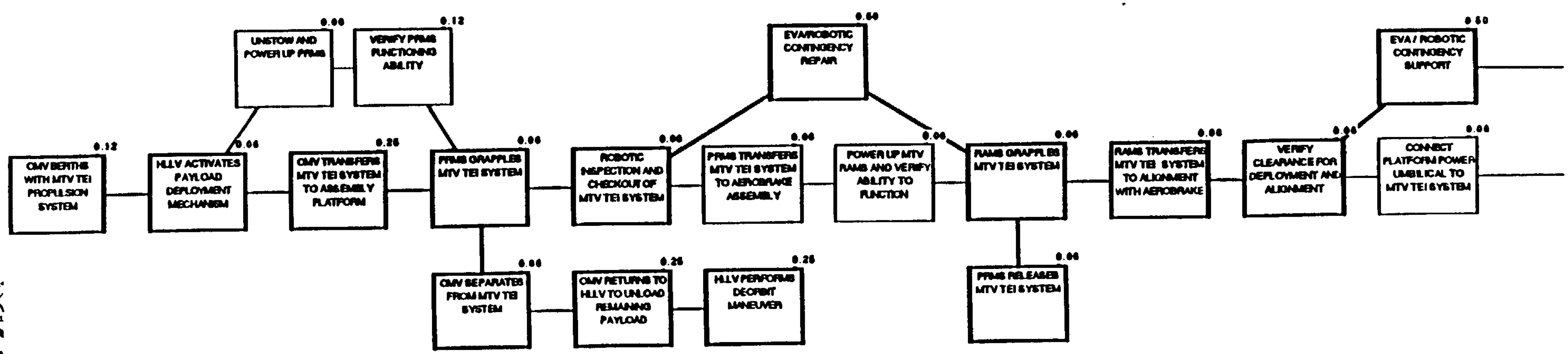

RAMS VERIFIES ALKINMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
RAMS MECHANICALLY FASTENS EIGHTH AND NINETH COMPONENT
0.10
PRMS PERFORMS COARSE ALIGNMENT AND FIXTURE OF TENTH COMPONENT FOR ASSEMBLY
RAMS VERIFIES ALIGNMENT AND PERFORMS FINE ADJUSTMENT AS REQUIRED
RAMS MECHANICALLY FASTENS NINETH AND TENTH COMPONENT
0.10
AEROBRAKE MECHANICAL ASSEMBLY COMPLETE
0.50
INSPECTION AND VERIFICATION OF MTV AEROBRAKE ASSEMBLY
0.12
RAISE AEROBRAKE ASSEMBLY WITH PAS TO ALLOW INSPECTION AND INSTALLATION OF TPS
PRMS INSPECTS TPS SYSTEM FOR ASSEMBLY DAMAGE
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
ROBOTIC INSPECTION AND CHECK OUT
EVA/ROBOTIC CONTINGENCY REPAIR AND ALIGNMENT
0.50
ASSEMBLY PLATFORM GRAPPLES COMPONENT WITH PRMS
PRMS TRANSFERS COMPONENT TO ASF STORAGE POSITION
HLLV ACTIVATES PAYLOAD DEPLOYMENT MECHANISM
OMV TRANSFERS MTV AEROBRAKE COMPONENT TO ASSEMBLY PLATFORM
0.25
OMV SEPARATES FROM AEROBRAKE COMPONENT
HLLV PERFORMS DEORBIT MANEUVER
0.25
OMV RETURNS TO FREEDOM
0.50

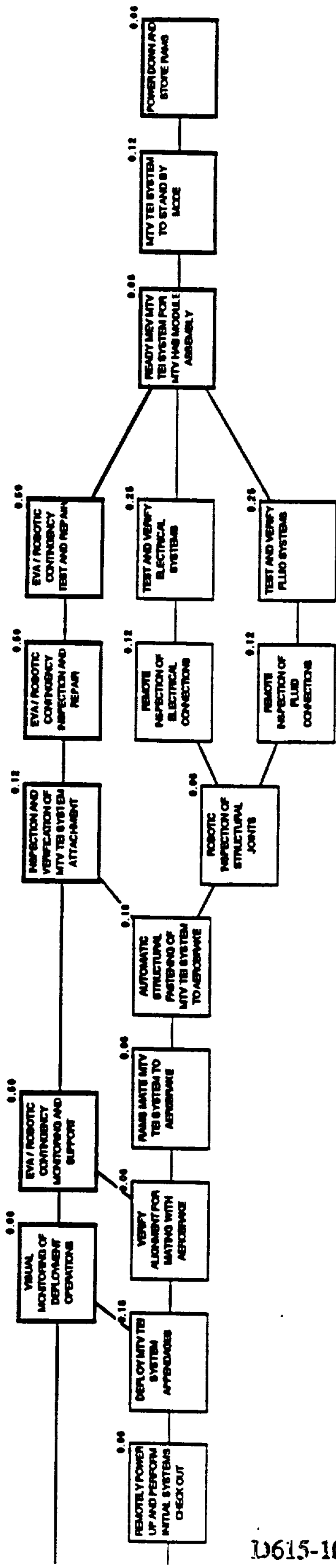

EVA/ROBOTIC CONTINGENCY REPAIR
PRMS INSTALLS TPS AT ASSEMBLY JOINTS
VERIFICATION OF TPS APPLICATION TO JOINTS OF AEROBRAKE
AEROBRAKE STORED FOR FUTURE ASSEMBLY OPERATIONS
POWER DOWN AND SECURE PLATFORM ASSEMBLY EQUIPMENT
PRMS POWERED DOWN AND STORED

HLLV MISSION SEVEN

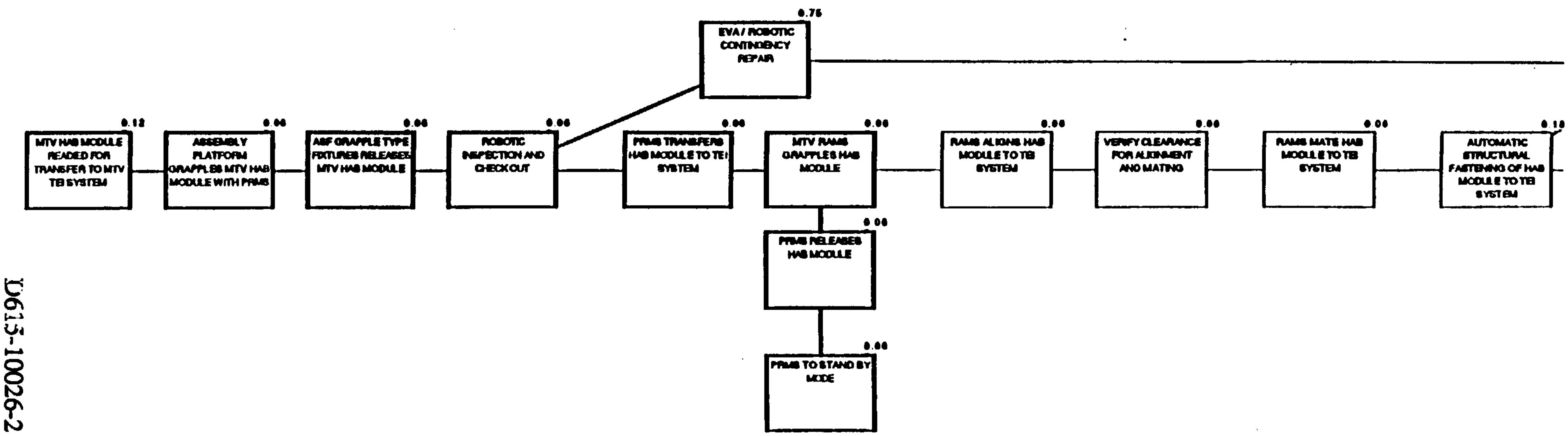


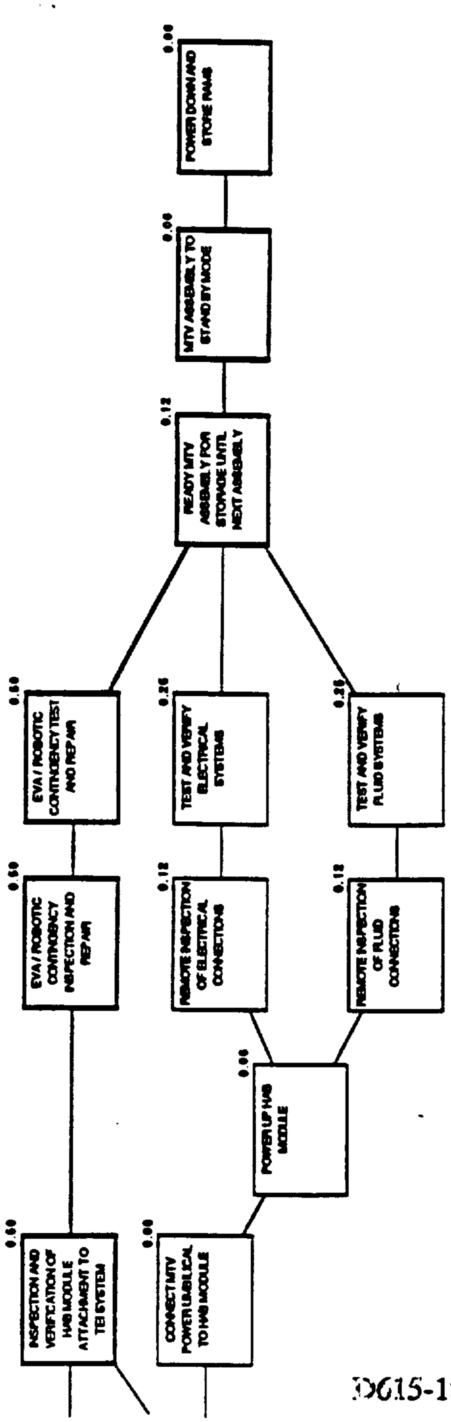
MTV TEI PROPULSION SYSTEM



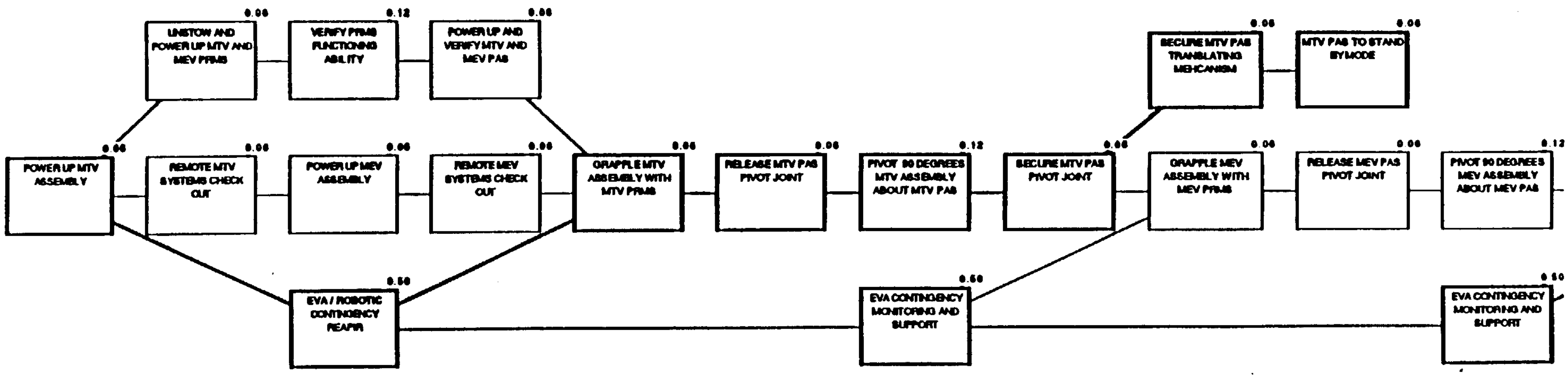

0.06
REMOTELY POWER UP AND PERFORM INITIAL SYSTEMS CHECK OUT
0.10
DEPLOY MTV TEI SYSTEM APPENDAGES
0.06
VISUAL MONITORING OF DEPLOYMENT OPERATIONS
0.06
VERIFY ALIGNMENT FOR MATING WITH AEROBRAKE
0.60
EVA / ROBOTIC CONTINGENCY MONITORING AND SUPPORT
0.06
RAMS MATE MTV TEI SYSTEM TO AEROBRAKE
0.10
AUTOMATIC STRUCTURAL FASTENING OF MTV TEI SYSTEM TO AEROBRAKE
0.12
INSPECTION AND VERIFICATION OF MTV TEI SYSTEM ATTACHMENT
0.06
ROBOTIC INSPECTION OF STRUCTURAL JOINTS
0.50
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
0.12
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
0.12
REMOTE INSPECTION OF FLUID CONNECTIONS
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.25
TEST AND VERIFY ELECTRICAL SYSTEMS
0.25
TEST AND VERIFY FLUID SYSTEMS
0.06
READY MEV MTV TEI SYSTEM FOR MTV HAB MODULE ASSEMBLY
0.12
MTV TEI SYSTEM TO STAND BY MODE
0.06
POWER DOWN AND STORE RAMS

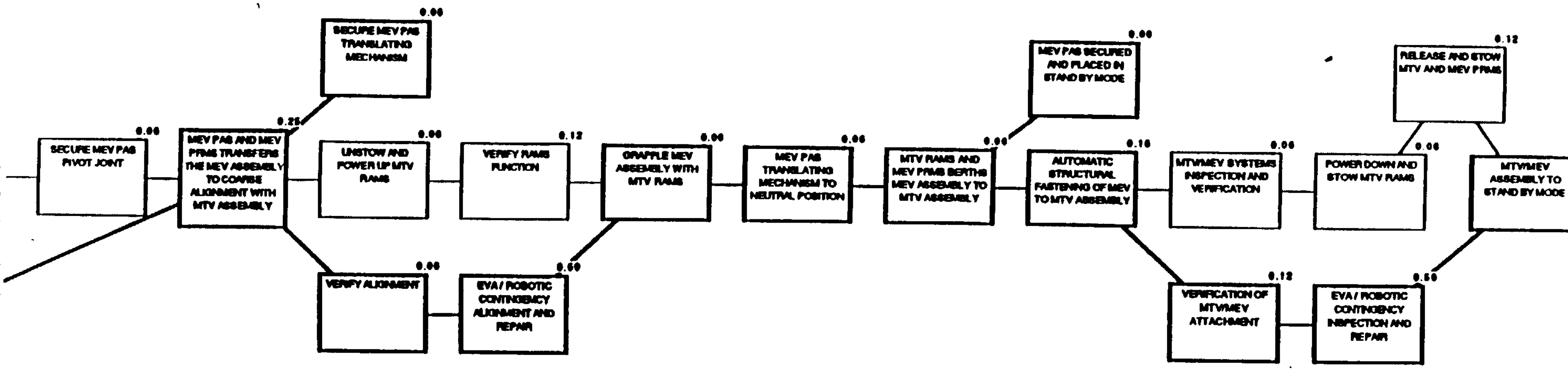
MTV ASSEMBLY



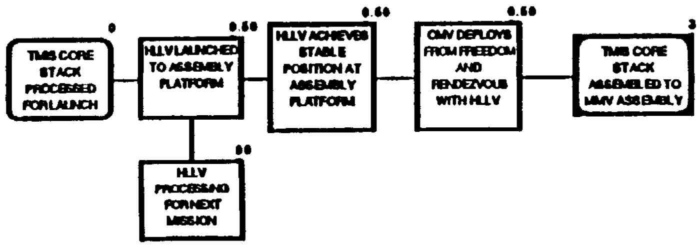

POWER DOWN AND STORE RAMS
MTV ASSEMBLY TO STAND BY MODE
READY MTV ASSEMBLY FOR STORAGE UNTIL NEXT ASSEMBLY
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
TEST AND VERIFY ELECTRICAL SYSTEMS
TEST AND VERIFY FLUID SYSTEMS
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
REMOTE INSPECTION OF ELECTRICAL CONNECTIONS
REMOTE INSPECTION OF FLUID CONNECTIONS
POWER UP HAB MODULE
INSPECTION AND VERIFICATION OF HAB MODULE ATTACHMENT TO TEI SYSTEM
CONNECT MTV POWER UMBILICAL TO HAB MODULE

MTV / MEV ASSEMBLY

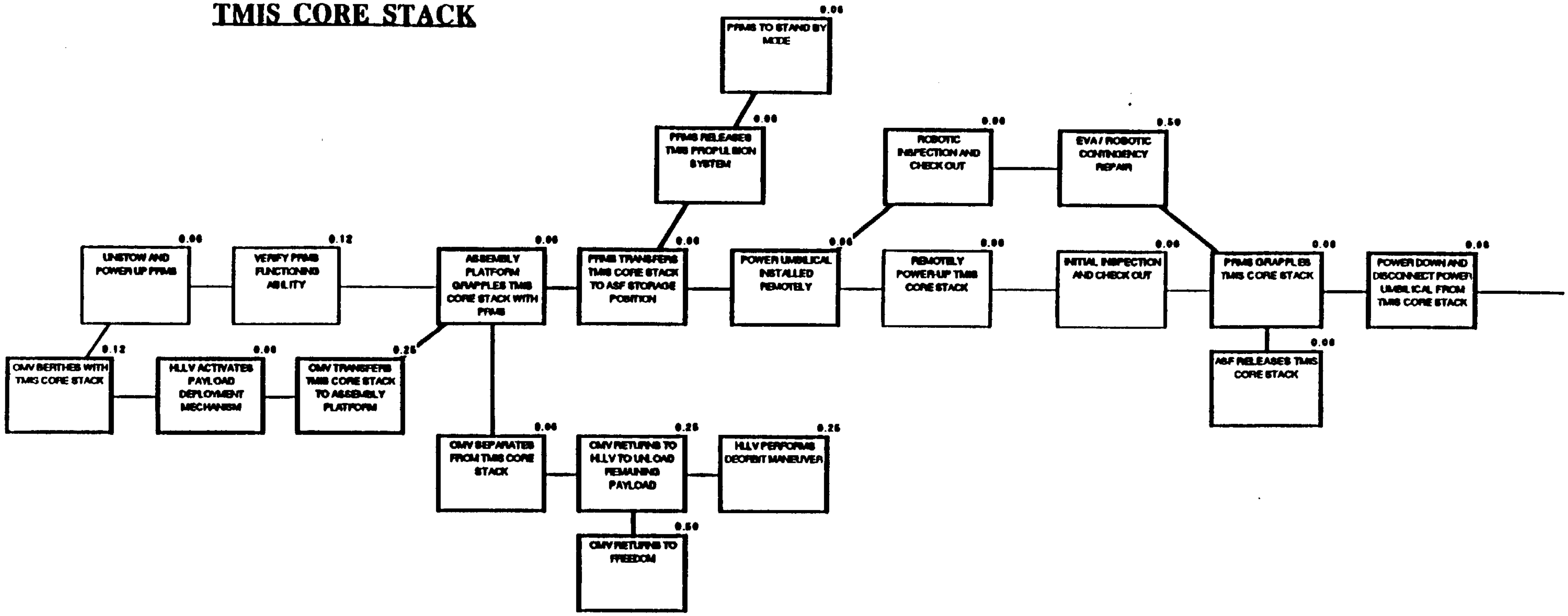


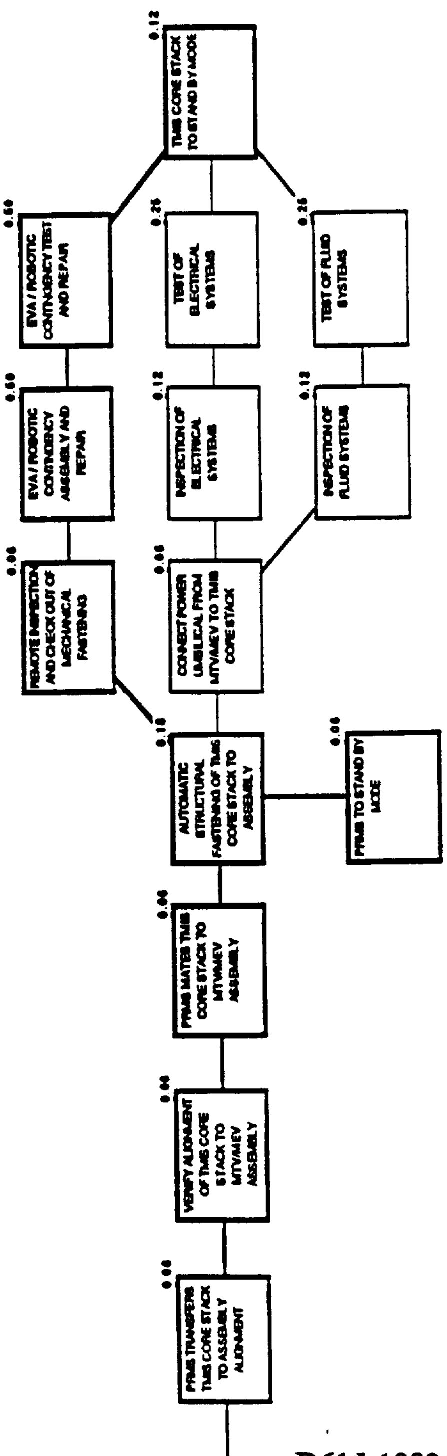

SECURE MEV PAS PIVOT JOINT
MEV PAS AND MEV PRMS TRANSFERS THE MEV ASSEMBLY TO COARSE ALIGNMENT WITH MTV ASSEMBLY
SECURE MEV PAS TRANSLATING MECHANISM
UNSTOW AND POWER UP MTV RAMS
VERIFY ALIGNMENT
VERIFY RAMS FUNCTION
EVA / ROBOTIC CONTINGENCY ALIGNMENT AND REPAIR
GRAPPLE MEV ASSEMBLY WITH MTV RAMS
MEV PAS TRANSLATING MECHANISM TO NEUTRAL POSITION
MTV RAMS AND MEV PRMS BERTHS MEV ASSEMBLY TO MTV ASSEMBLY
MEV PAS SECURED AND PLACED IN STAND BY MODE
AUTOMATIC STRUCTURAL FASTENING OF MEV TO MTV ASSEMBLY
MTV/MEV SYSTEMS INSPECTION AND VERIFICATION
VERIFICATION OF MTV/MEV ATTACHMENT
POWER DOWN AND STOW MTV RAMS
RELEASE AND STOW MTV AND MEV PRMS
EVA / ROBOTIC CONTINGENCY INSPECTION AND REPAIR
MTV/MEV ASSEMBLY TO STAND BY MODE

HLLV MISSION EIGHT

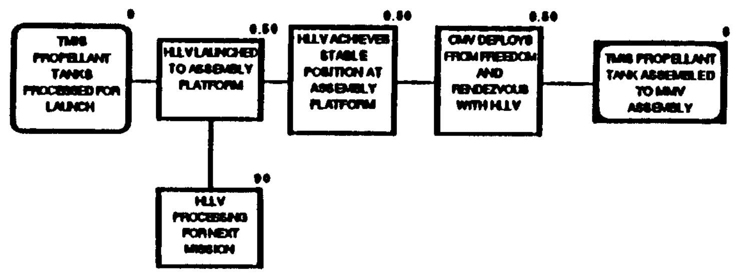


TMIS CORE STACK

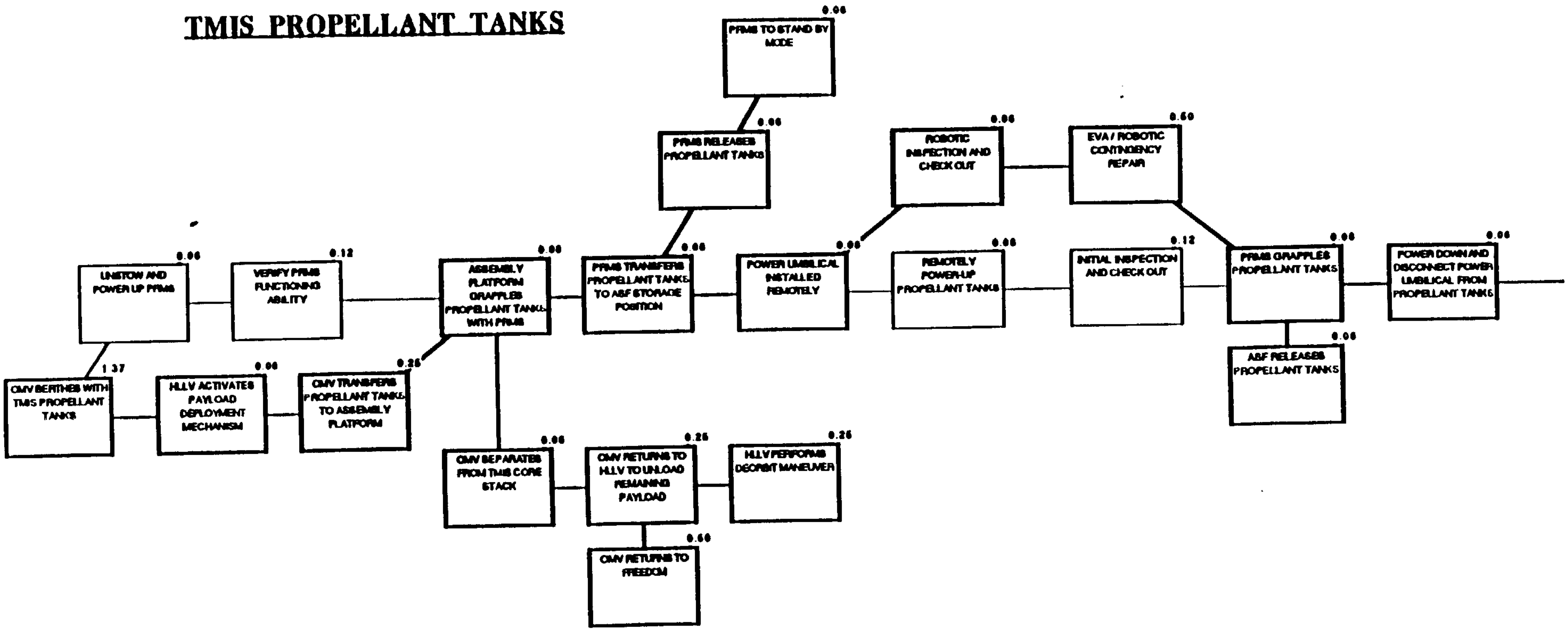


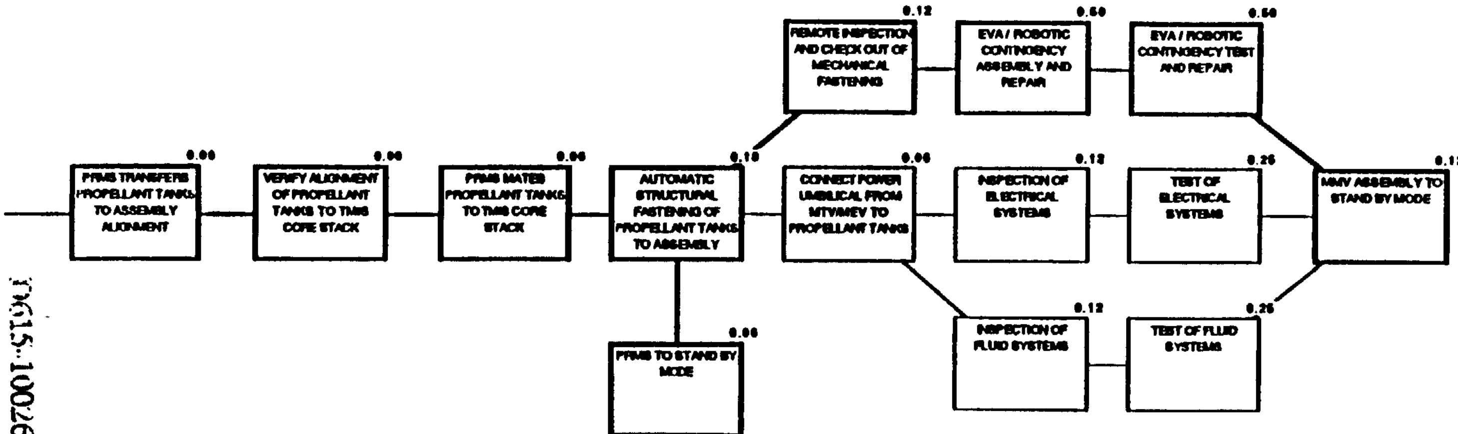

PRMS TRANSFERS TMIS CORE STACK TO ASSEMBLY ALIGNMENT
0.06
VERIFY ALIGNMENT OF TMIS CORE STACK TO MTV/MEV ASSEMBLY
0.06
PRMS MATES TMIS CORE STACK TO MTV/MEV ASSEMBLY
0.06
AUTOMATIC STRUCTURAL FASTENING OF TMIS CORE STACK TO ASSEMBLY
0.18
PRMS TO STAND BY MODE
0.06
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.06
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.50
CONNECT POWER UMBILICAL FROM MTV/MEV TO TMIS CORE STACK
0.06
INSPECTION OF ELECTRICAL SYSTEMS
0.12
TEST OF ELECTRICAL SYSTEMS
0.25
INSPECTION OF FLUID SYSTEMS
0.12
TEST OF FLUID SYSTEMS
0.25
TMIS CORE STACK TO STAND BY MODE
0.12

HLLV MISSION NINE

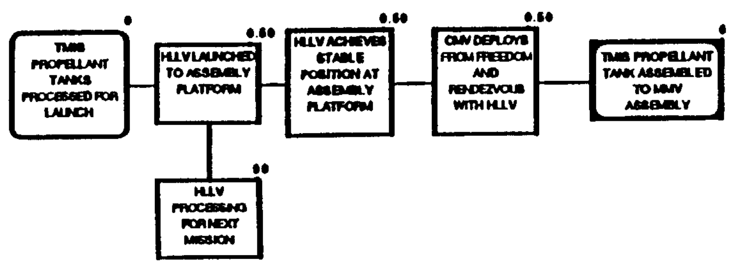


TMIS PROPELLANT TANKS

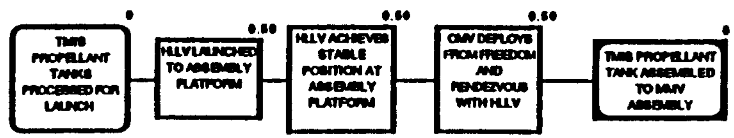



PRMS TRANSFERS PROPELLANT TANKS TO ASSEMBLY ALIGNMENT
0.06
VERIFY ALIGNMENT OF PROPELLANT TANKS TO TMIS CORE STACK
0.06
PRMS MATES PROPELLANT TANKS TO TMIS CORE STACK
0.06
AUTOMATIC STRUCTURAL FASTENING OF PROPELLANT TANKS TO ASSEMBLY
0.18
PRMS TO STAND BY MODE
0.06
REMOTE INSPECTION AND CHECK OUT OF MECHANICAL FASTENING
0.12
EVA / ROBOTIC CONTINGENCY ASSEMBLY AND REPAIR
0.50
EVA / ROBOTIC CONTINGENCY TEST AND REPAIR
0.50
CONNECT POWER UMBILICAL FROM MTV/MEV TO PROPELLANT TANKS
0.06
INSPECTION OF ELECTRICAL SYSTEMS
0.12
TEST OF ELECTRICAL SYSTEMS
0.25
INSPECTION OF FLUID SYSTEMS
0.12
TEST OF FLUID SYSTEMS
0.25
MMV ASSEMBLY TO STAND BY MODE
0.12

HLLV MISSION TEN



HLLV MISSION ELEVEN



On-Orbit Support Equipment

- Assembly scenario is to complete major assemblies robotically with crew support for contingency only.
- Robotic operations will be controlled by Ground Control Center primarily, with control being "handed off" to assembly crew during contingency operations.
- **Platform Remote Manipulating System (PRMS)**
  - Two complete systems required for MMV assembly
  - Each main arm can span the entire 30M diameter of the aerobrake
  - Each main arm has a "hold down" grapple to secure the working end to EVA handrails
  - Each main arm has a 2.5M work arm capable of precise movements and operations
  - Elbow joints feature n-pi rotational freedom and the wrist joints are compact roll-pitch-roll units
  - Video cameras allow direct monitoring and machine vision from the end effector
  - All hardware is bar-coded for positive machine recognition
  - End effector is equipped with a 6-axis EM antennae, which determine location and orientation relative to EM beacons distributed across the assembly site.
  - Tools and hardware required for assembly operations will be secured to the main arm, within reaching distance of the work area
  - Each arm will be capable of maneuvering 128 metric tons (proposed mobile servicing center 10-12-89)
  - Each arm will be track-mounted so as to maneuver about the perimeter of the assembly area

On-Orbit Support Equipment (cont'd)

- **Remote Aerobrake Manipulating System (RAMS)**
  - Same characteristics as PRMS for commonality
  - 2 systems attached to track on each aerobrake

- **Platform Anchor System (PAS)**
  - 4 units required for assembly
  - Track-mounted to allow movement during assembly operations
  - Extendable to TBD height to allow access for TPS installation and inspection
  - Lift capabilities of 128 metric tons
  - Grapple-type end effector to anchor components to platform
  - Elbow joints with 0.50pi rotational freedom
  - Wrist joints with roll-pitch-roll movements

- **Assembly Support Fixture (ASF)**
  - Fixed storage locations. TBD units required for assembly across the assembly platform
  - Removable grapple-type end effector
  - Able to support up to 128 metric tons
  - Grapple fitting remotely controlled to release and secure components

- **Lighting & Video Monitoring**
  - PRMS and RAMS assembly arms will have required lighting and video/fiber optic monitoring capabilities.
  - Portable lighting will be available as required

- **EVA Handrails and Tether Tie-Down Points**
  - Available on each assembly component

On-Orbit Support Equipment (cont'd)

- **Electrical Power**
  - Will be supplied by assembly platform (Solar Dynamics)
- **MTV Habitat Module**
  - Provide crew stationing facility
- **SSF Type Node**
  - Houses local control of MAP and Assembly Equipment
  - Provide berthing port for Logistics Module
  - Provide berthing port for ACRV
- **SSF-Type Logistics Module**
  - Provide consumables storage and transportation

Issues and Concerns

- TPS Installation and Inspection
- Launch Vehicle Intergration
- Robotic Operations
- Mechanical Fasteners vs. Welding

Revised On-Orbit Analysis

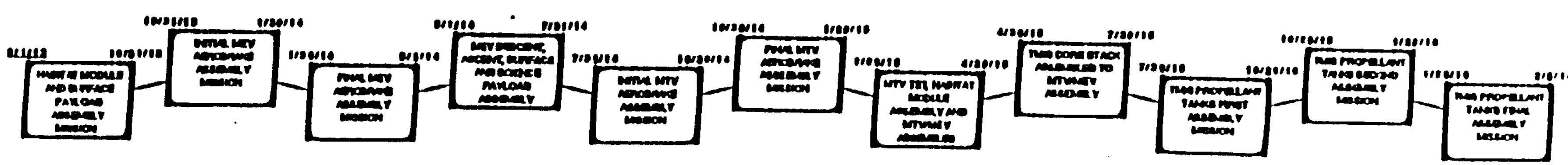
The following two charts summarize the results of the Revised On-Orbit Assembly Analysis. Revisions were made because the analysis for HLLV processing was found to be in series to On-Orbit Assembly, where as the analysis should have been in parallel.

/STCAEM/dks/12June90

On-Orbit Assembly Analysis

- Assembly Analysis Presented at Second Quarter Review has Been Revised
- Revised Data shows 5 months less time required for On-Orbit Assembly
  - Original HLLV ground processing time was calculated in series with the On-Orbit Assembly time
  - Revised analysis calculates the HLLV ground processing time in parallel with the On-Orbit Assembly time
- Original On-Orbit Assembly Start date of January 2014
- Original On-Orbit Assembly Completion date of December 2016
- Revised On-Orbit Assembly Start date of August 2013
- Revised On-Orbit Assembly Completion date of February 2016

REVISED ASSEMBLY SCHEDULE



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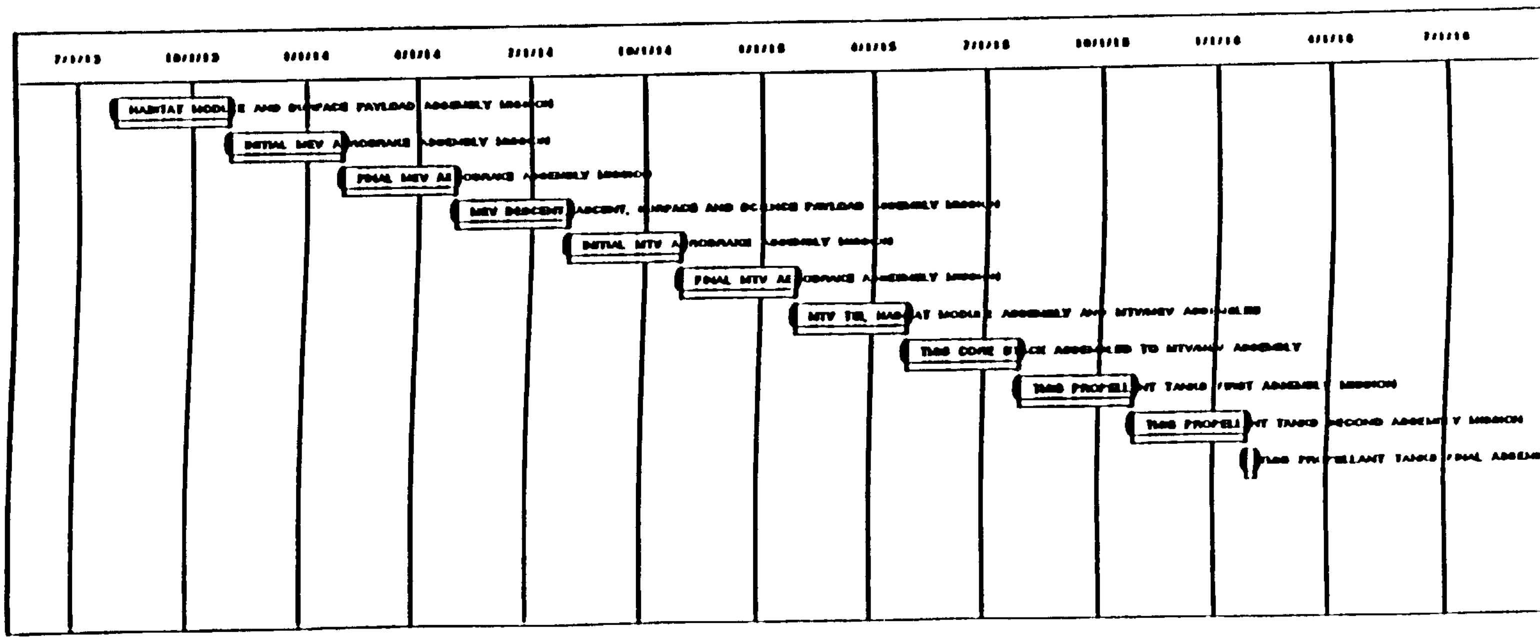
On-Orbit Assembly Analysis

Name	Earliest Start	Earliest Finish	Subproject	Days
HABITAT MODULE AND SURFACE PAYLOAD ASSEMBLY MISSION	8/1/13	10/31/13	HLLV MISSION ONE	91
INITIAL MEV AEROBRAKE ASSEMBLY MISSION	10/31/13	1/30/14	HLLV MISSION TWO	91
FINAL MEV AEROBRAKE ASSEMBLY MISSION	1/30/14	5/1/14	HLLV MISSION THREE	91
MEV DESCENT, ASCENT, SURFACE AND SCIENCE PAYLOAD ASSEMBLY	5/1/14	7/31/14	HLLV MISSION FOUR	91
INITIAL MTV AEROBRAKE ASSEMBLY MISSION	7/31/14	10/30/14	HLLV MISSION FIVE	91
FINAL MTV AEROBRAKE ASSEEMBLY MISSION	10/30/14	1/29/15	HLLV MISSION SIX	91
MTV TEI, HABITAT MODULE ASSEMBLY AND MTV/MEV ASSEMBLED	1/29/15	4/30/15	HLLV MISSION SEVEN	91
TMIS CORE STACK ASSEMBLED TO MTV/MEV ASSEMBLY	4/30/15	7/30/15	HLLV MISSION EIGHT	91
TMIS PROPELLANT TANKS FIRST ASSEMBLY MISSION	7/30/15	10/29/15	HLLV MISSION NINE	91
TMIS PROPELLANT TANKS SECOND ASSEMBLY MISSION	10/29/15	1/28/16	HLLV MISSION TEN	91
TMIS PROPELLANT TANKS FINAL ASSEMBLY MISSION	1/28/16	2/5/16	HLLV MISSION ELEVEN	8



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Node Purpose & Minimal Requirements

Purpose

- Integration
- De-integration
- Re-integration

of mission vehicles, including assembly, processing, resupply and refurbishment

Top-level Requirements

- Accessibility
- Provision of support services to mission vehicle

Functional Approach

- Analyze specific functions necessary to provide required services to the mission vehicles
- Identify synergistic ways of providing those functions, emphasizing "operational" solutions (e.g. using/proving onboard vehicle systems, resupplying before mission departure)
- Defer device-driven solutions until minimum common requirements are distilled, which cannot be satisfied by hardware already "procured" for the vehicle itself

STCAEM/bs/28March90

Assembly Node Purpose and Requirements

Assembly Node Purpose and Minimal Requirements was developed to support a study that MSFC was performing to determine the advantages and disadvantages of different Orbits for an Assembly Node. Each vehicle option has advantages for being assembled and launched from different Orbits. The data shown in the following charts is for the Cryo/Aerobrake Vehicle.

/STCAEM/dks/12June90

Node Comparisons

Node / Data to Mars	LEO MASE REF.	GEO Case 1	GEO* Case 3	LLO*	L2*
Δ V to Mars (m/sec)	4281	5402	2423	1995	1315
TMI S mass	470.3 t	751.5 t	191.8 t	147.7 t	87.9 t
Debris Env.(total time)	Continuous	0	< 1 hr	< 1 hr	< 1 hr
Radiation Env. total time in Van Allen belts	~2hrs	0	< 2 hrs	< 2 hrs	< 2 hrs
Radiation Environment	Trapped, SAA	Trapped, GCR, SPE	Trapped, GCR, SPE	GCR, SPE	GCR, SPE
Crew to Node ΔV /t	SSF*** ΔV=10-100m/sec hrs-days	SSF - 6 hrs ΔV= 4200m/sec	SSF - 6 hrs ΔV=4200m/sec	SSF- 3-13days ΔV= 4000 m/s Moon- 2hr ΔV= 2100m/s	SSF -8-18 days ΔV= 3374m/s Moon -3 days ΔV= 2900m/s
Launch window timing	e.s o.** ~ 10 min orbit align ~ 5d opps/day ~15-16 rec = 30-60 days	e.s.o.~ 1/day orbit align ~ planetary position opp ops/day = continous		e.s.o. @ 10 min alignment =12hr rec.= 27 days retrograde opp.	alignment =12hr rec.= 27 days retrograde opp.
Logistics wndow timing	anytime	5 hr transit, 2 opportunities/day		every 10 days	transit from Moon = 3 day

* denotes PEGA (powered Earth gravity assist)
** engine start opportunity
*** co- orbiting with SSF costs ΔV to maintain poisition relative to SSF(not continous thrust)

GCR = Galatic Cosmic Radiation
SPE = Solar Proton Events

./STCAEM/pb/17April90

Assembly Node Concepts Pros and Cons

Node Concepts	Key Features/Advantages	Key Disadvantages
Dedicated Assembly Node	• Abundant storage • Totally self-contained • Vehicle systems unused • Multiple robot arms • Sections of vehicle may be assembled simultaneously	• Larger than SSF • Will take long time to construct • Excessive reboost requirements • Mechanically complex • Local debris shielding required • Must be in place prior to vehicle assembly
I-Beam Platform	• Can be carried up in first HLLVflight • Can easily reach most parts of vehicle with two robot arms • Uses vehicle for comm., data, RCS, power after initial deployment • Can serve as base for experiments	• Fuel cells, batteries required for initial deployment • Limited storage area • Precursor mission required for deployment
"Smart" HLLV Platform	• No additional platform required • HLLV shroud provides limited debris shielding • HLLV provides for communication, data, RCS, GNC, etc. • Robot arms transferable to NTR	• Increased HLLV complexity • Reboost fuel has to be replenished • Limited storage • Vehicle must be detached from HLLV prior to assembly complete • Local debris shielding required
Hinged Truss Platform	• Uses vehicle truss as assembly platform; no other platform needed • Reach to remote engine section of vehicle provided by flexing truss at hinges • Vehicle subsystems used; no additional systems necessary	• Requires a precursor mission to deploy truss • Batteries, fuel cells necessary for initial deployment • Reboost, comm., data, power, must be in place prior to assembly start • Limited storage • Local debris shielding required
Vehicle as its own Platform	• Reduces needed on-orbit infrastructure • Deletes additional facilities and resources needed for designing, builiding, launching, and maintaining separate assembly platform	• Requires dedicated HLLV flight for non-optimized packaged first element • Requires vehicle to have additional control, reboost • No additional storage • Requires batteries or fuel cells for initial deployment • Requires localized debris shielding

Assembly Node Concepts Pros and Cons
(continued)

Node Concepts	Key Features/Advantages	Key Disadvantages
Assembly Flyer Platform	• Performs HLLV unloading, payload/crew transport, and assembly with one vehicle • Compatible with SSF • Capable of manned/robotic operations • Uses CTV for main P/A • Can serve as free flying platform between assemblies	• No additional storage • Requires vehicle to have additional control and reboost systems • Requires development and production of sophisticated man-rated space vehicle • Requires localized debris shielding
SSF Based Assembly of First Element	• Uses planned SSF growth concept • Provides quick and easy crew logistics access to initial assembly operations • Allows verification and checkout of critical systems prior to independent vehicle operations • Does not disrupt SSF operations beyond first assembly mission (remainder of assembly based from vehicle itself)	• Impact to SSF (resources, microgravity, drag, etc.) • Eventually requires vehicle to have additional control and reboost systems • Requires localized debris shielding • No additional storage beyond first element
Tethered off-SSF Assembly Platform	• Compatible with current SSF design • Provides quick and easy crew and logistics access to entire assembly and propellant transfer operations • Microgravity and dynamic loads impacts to SSF minimized by tether • Removes hazardous operations and materials to SSF standoff distance	• Impact to SSF resources • Requires localized debris shielding • No additional storage • Requires additional reboost and control systems on SSF

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Ground

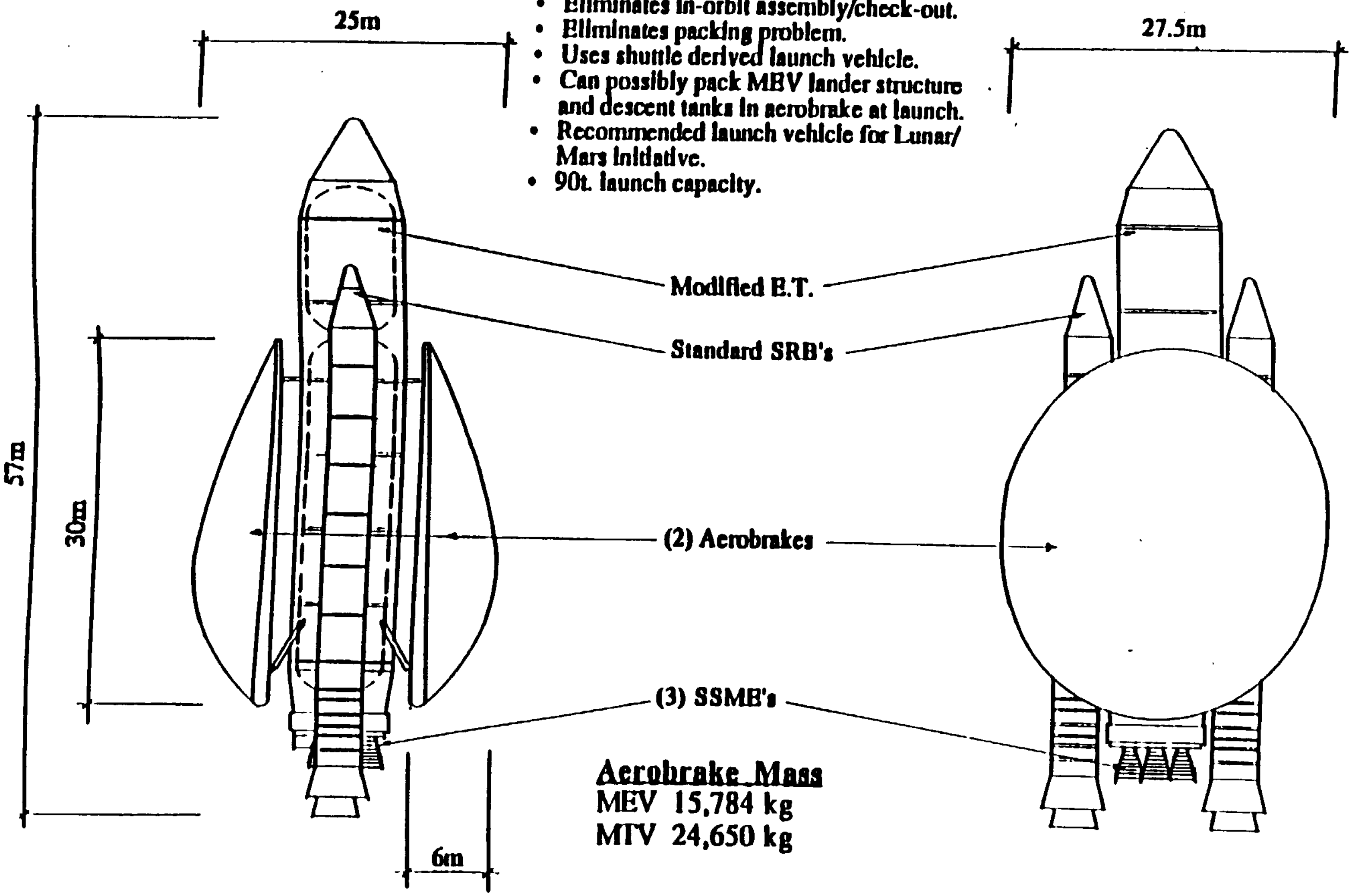
Integrated Aerobrake Launch Option

On-Orbit Assembly of the Mars Aerobrake(s) require two 10.5 meter dia HLLV launches each and 180 days (due to 90 day ground processing time required for the HLLV). A concept which would deliver an assembled aerobrake or aerobrakes to LEO is shown in the following chart. This concept utilizes a Shuttle derived In-Line vehicle to launch two aerobrakes to LEO.

STCAEM/dks/12June90

Shuttle Derived Aerobrake Launch Option

- Aerobrake completely assembled on Earth.
- Eliminates in-orbit assembly/check-out.
- Eliminates packing problem.
- Uses shuttle derived launch vehicle.
- Can possibly pack MEV lander structure and descent tanks in aerobrake at launch.
- Recommended launch vehicle for Lunar/Mars initiative.
- 90t. launch capacity.



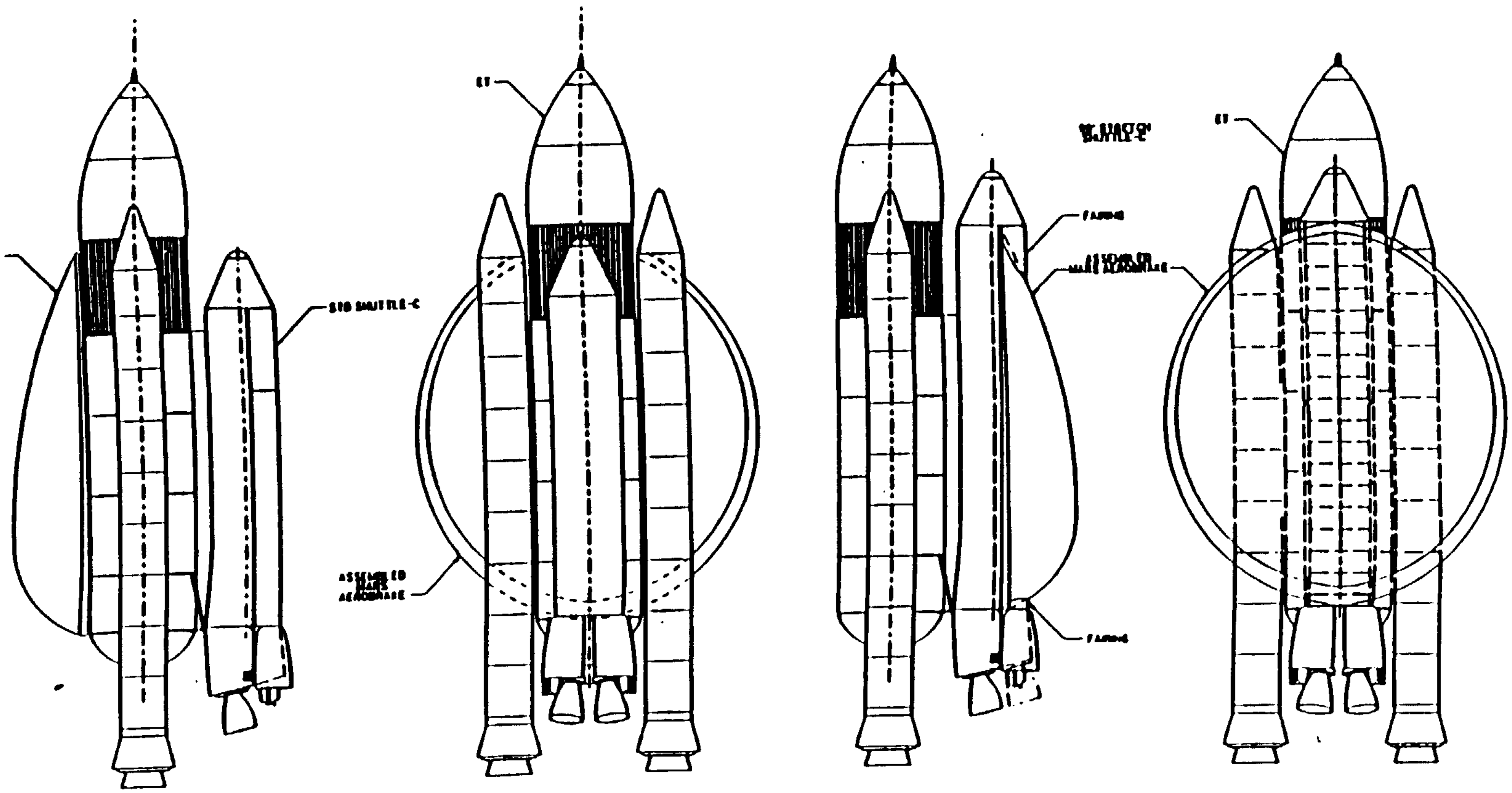
Aerobrake Mass
MEV 15,784 kg
MTV 24,650 kg

/STCAEM/dks/5June90

Integrated Aerobrake Launch Option

The following chart shows a Shuttle-C / Aerobrake Integrated Launch Option. This option would launch a single Aerobrake to LEO along with other payload stored in the Shuttle-C Payload Shroud.

Shuttle - C Aerobrake Launch Options



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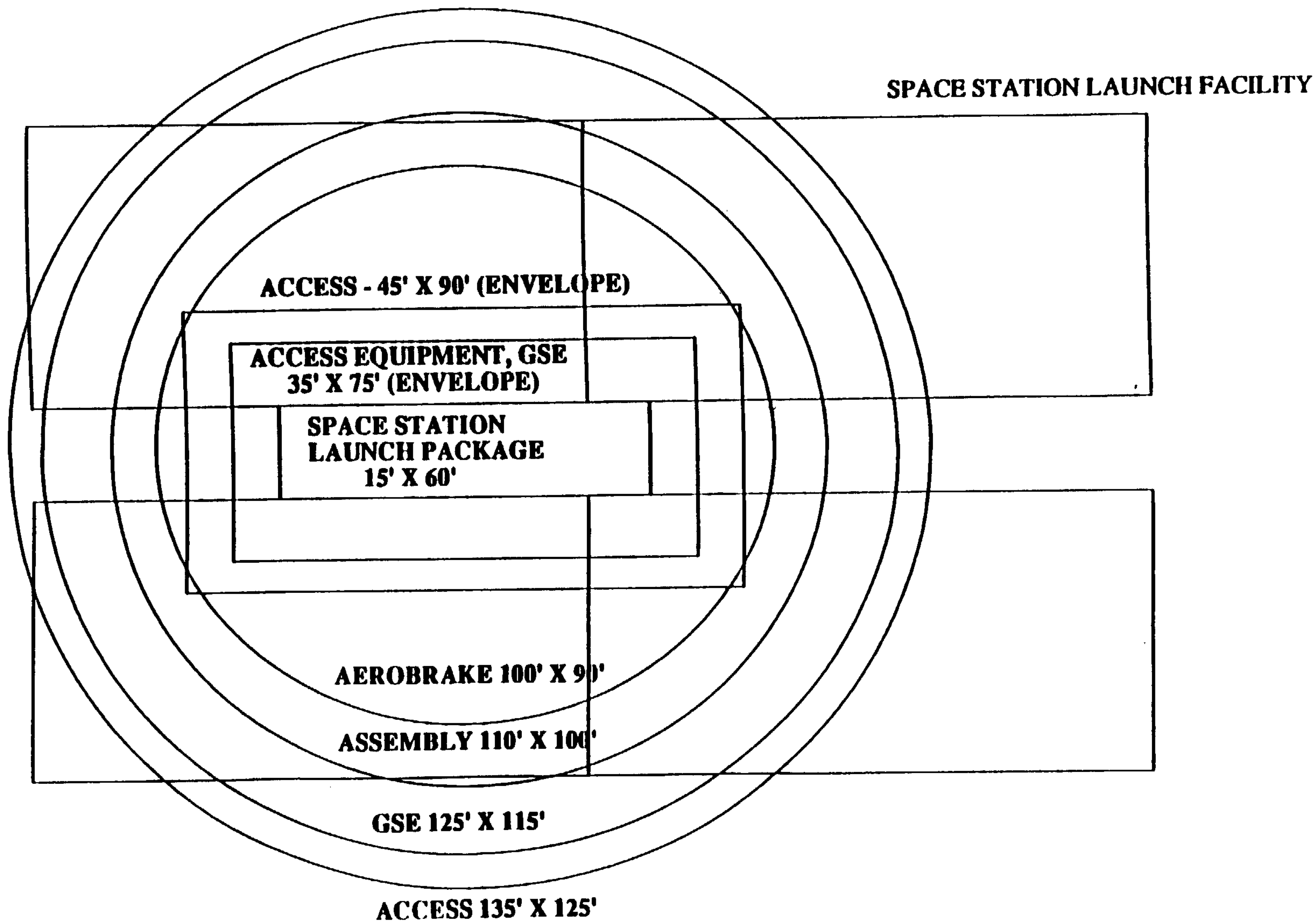
Size Comparison SSF Launch Facility With Aerobrake Footprint

This is a diagram of the low L/D aerobrake with its requiremented access corridor, space for general support equipment (GSE) and assembly activities superimposed on the footprint of a space station payload, and over four adjoining SSF payload footprints. The Space Station facility working area is 5 of the SSF payload footprints, it will be to small for aerobrake assembly and manipulation. A new facility will be needed.

Size Comparison SSF Launch Facility With Aerobrake Footprint



Mars Aerobrake Assembly and Integration at the VAB

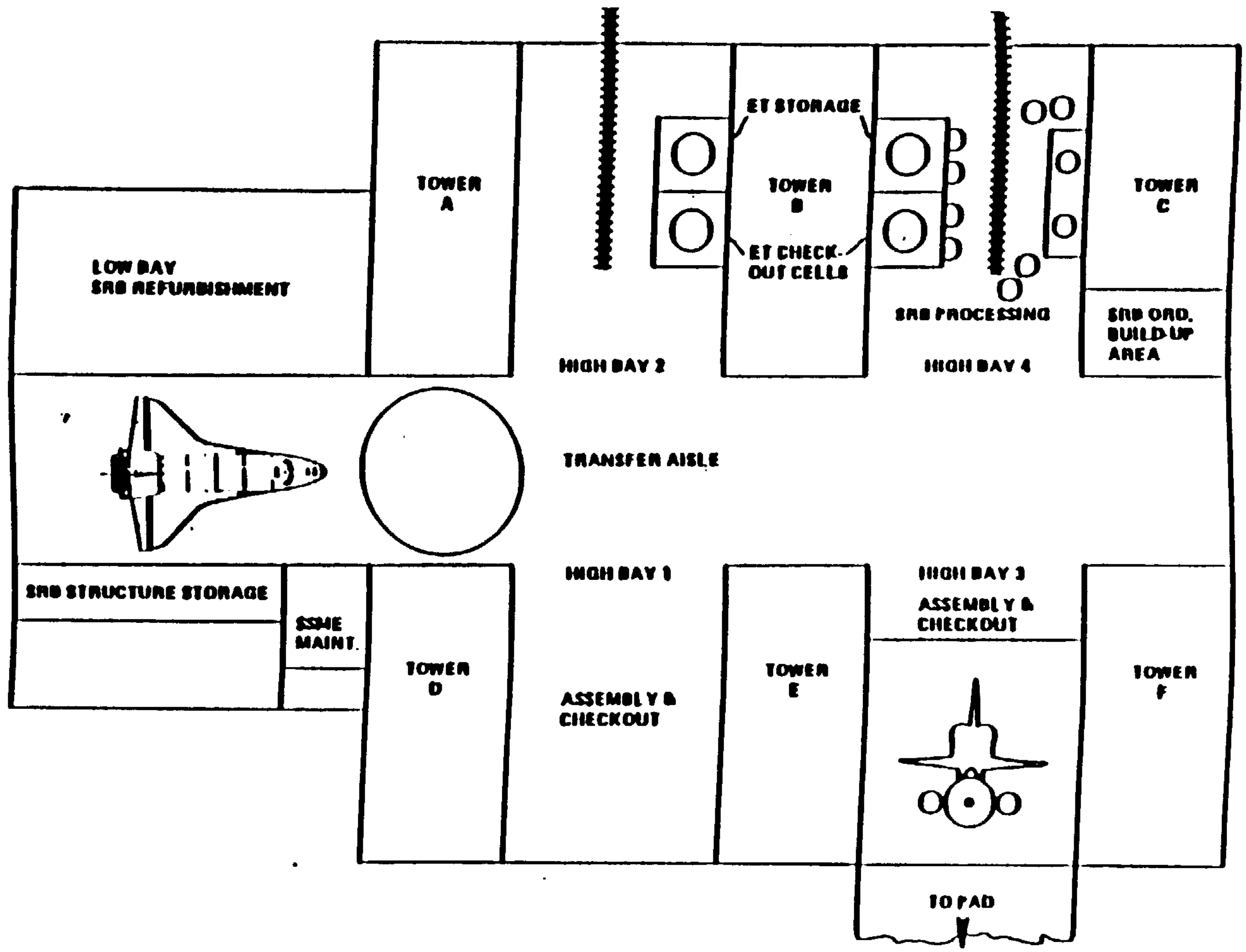
The current Vehicle Assembly Building (VAB) has a transfer aisle with the dimensions of 92.5 feet* the aerobrake is 91.8 feet,* as shown in the following chart, the Aerobrake can be transferred through the transfer aisle horizontally.

* 28.2 meters

* 28 meters

/STCAEM/dks/12June90

Mars Aerobrake Assembly at VAB



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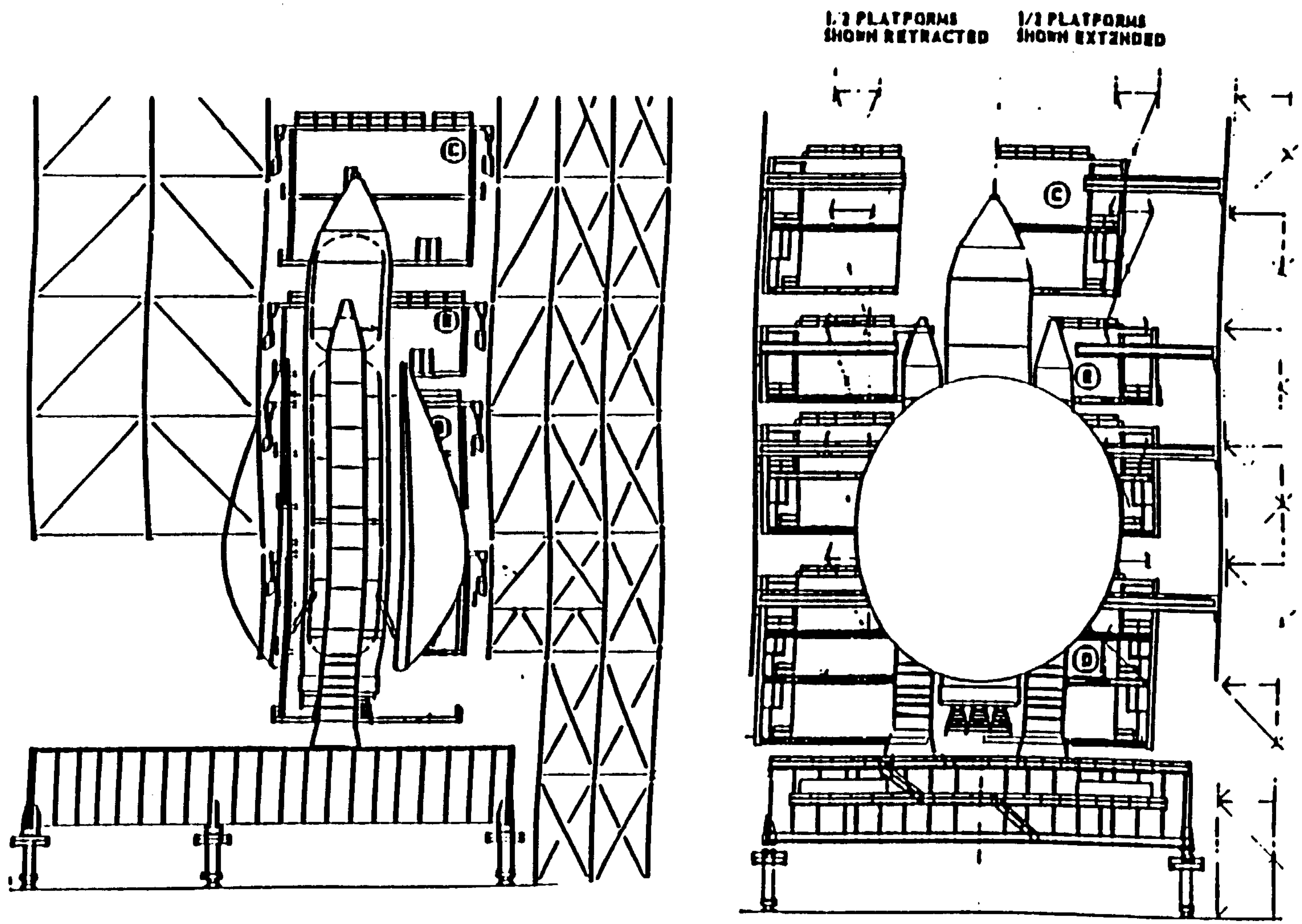
Mars Aerobrake Launch Option

Modifications of existing equipment will be required to assemble and check out the Integrated Launch Option in the current VAB High Bay.

Aerobrake Launch Option



VAB High Bay

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Launch Site Impacts

The following Launch Site Impacts were derived from current facility and equipment limitations. Refurbishment and modification of existing equipment or construction of new facilities will accommodate an Integrated Launch Option.

Launch Vehicle / Integral Aerobrake
Launch Site Impacts

- Transporter required for fully assembled Aerobrake
- VAB High Bay access platforms will require modifications
- Aerobrake will prohibit use of the Rotating Service Structure without major modifications
- Fixed Service Structure swing arm extension and retraction may interfer with the Aerobrake
- Large Aerobrake cross-sectional area will impart large wind loads to the launch vehicle
  - Increased loads to hold down fixtures
  - Revised launch commit criteria for maximum winds at launch

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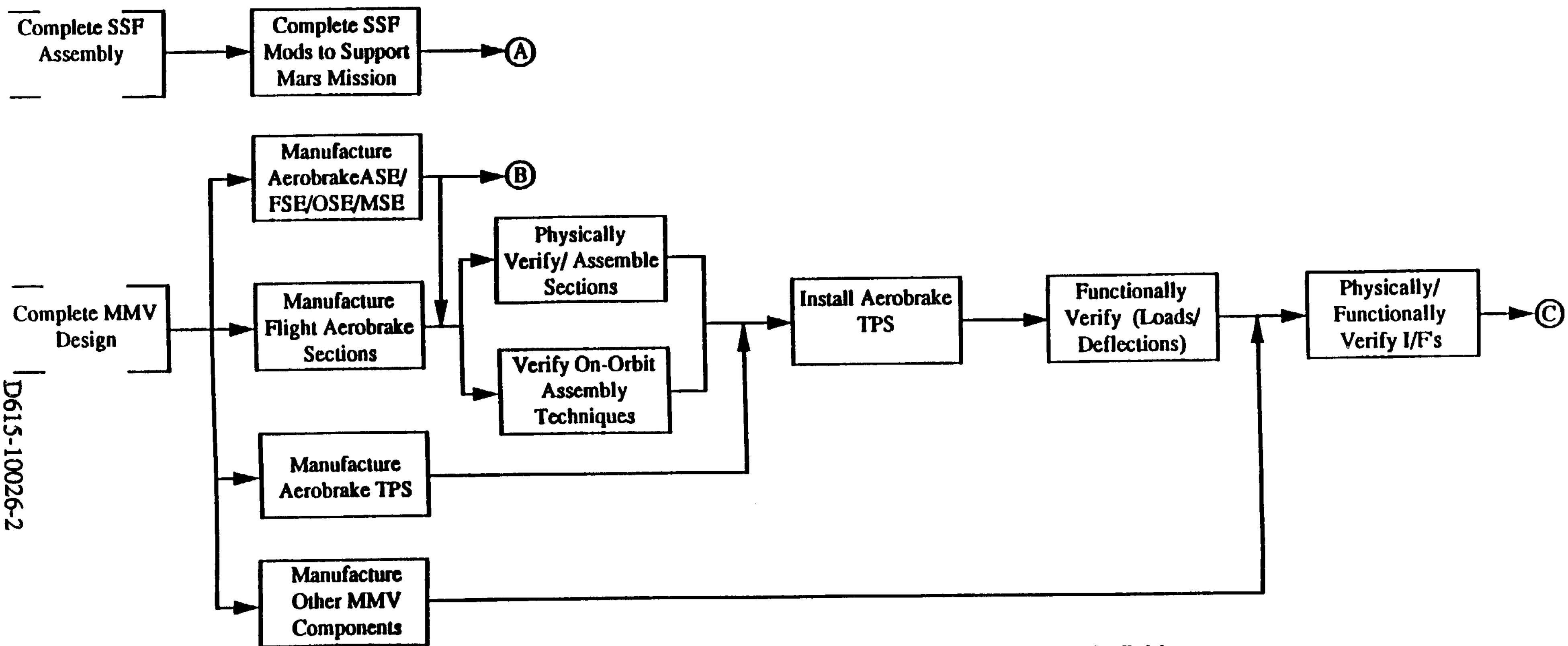
Aerobrake Preflight Operations

We have began analyzing the necessary functions that must be preformed to the aerobrake, the largest, most fragile and difficult to launch, piece of spacecraft. This chart and the following one give the analyses of functional flows for ground operation and requirements.

Aerobrake Preflight Operations



Definitions:

ASE (Airborne Support Equipment)
Stays in Launch Vehicle
FSE (Flight Support Equipment)
Used to Package MMV Components
(Struts Separating Aerobrake Sections)
OSE (Orbital Support Equipment)
Used During Assembly But Doesn't
go to Mars
MSE (Mission Support Equipment)
Used During Assembly and Goes to Mars

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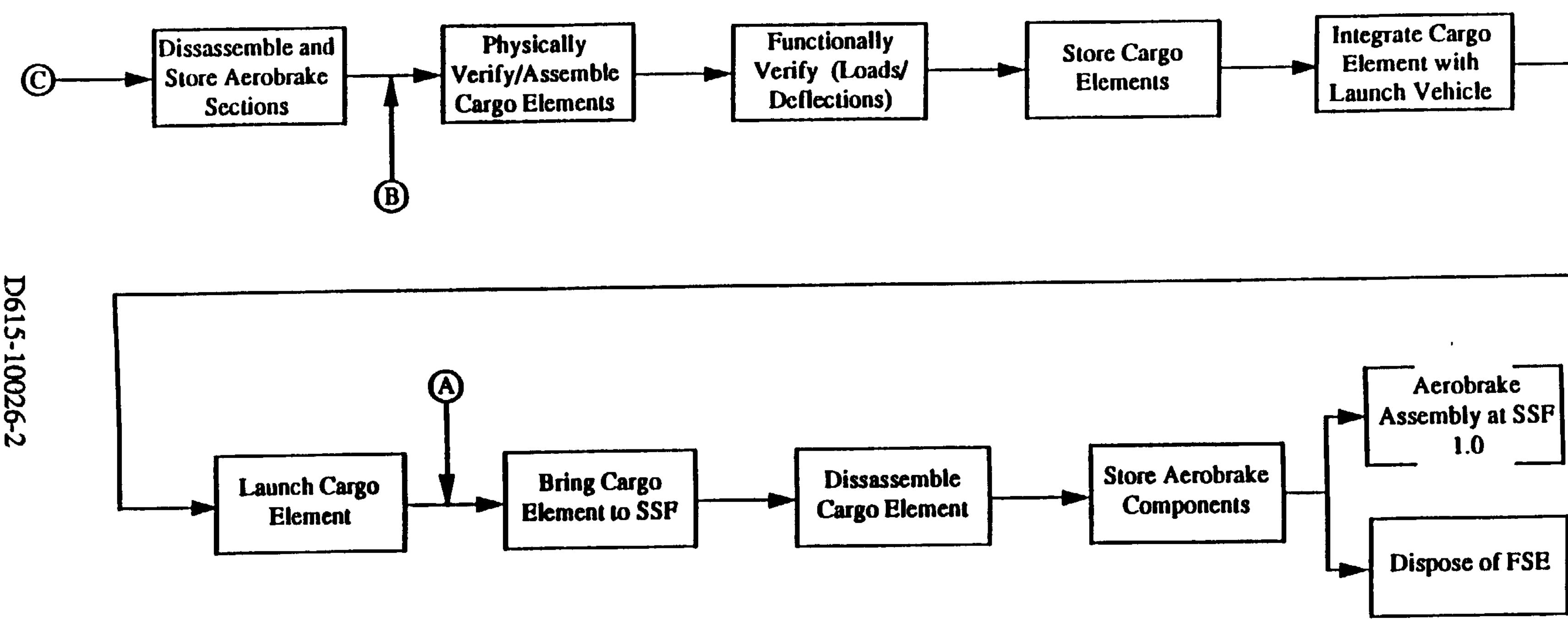
Aerobrake Preflight Operations (Cont)



Preflight Ops/13/12-4-89/Cox

Ground Assembly and Check Out

The following chart shows the Ground Rules and Assumptions developed for Ground Assembly Analysis of the Cryo/Aerobrake Vehicle.

/STCAEM/dks/12June90

Ground Rules / Assumptions For Ground Assembly

- A System is a group of components and supporting structure that is integrated by a contractor and delivered as a unit to the processing facility (eg. MEV Aerobrake, MEV Descent Lander, Ascent System, etc.).
- System Interfaces are those which transmit data, power, or fluids across the system's boundaries and mechanically secure one system to another.
- Subsystems Interfaces are those which are internal to a System.
- Subsystem Interfaces are verified by the manufacturer prior to System integration.
- Component Interfaces are those which are internal to a Subsystem.
- Component Interfaces are verified by the manufacturer during Subsystem Assembly.
- Interfaces verified prior to System Level Integration will be accepted with no repetition of tests.
- Flight Hardware will be used to verify System Interfaces.
- Ground facilities will simulate assembly node operations and limitations.

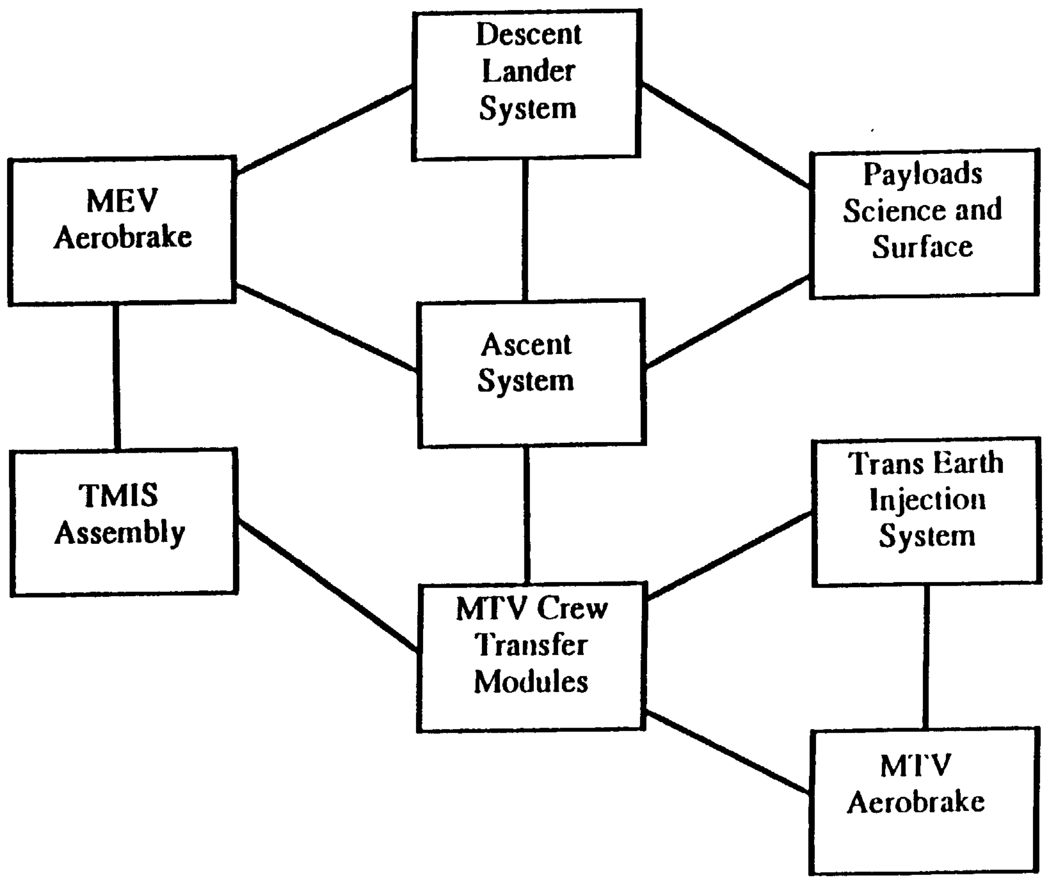
/STCAEM/dks/20March90

MMV Top Level System Interfaces

The following chart shows the Top Level System Interfaces of the Cryo/Aerobrake Vehicle.

/STCAEM/dks/12June90

MMV System Interfaces



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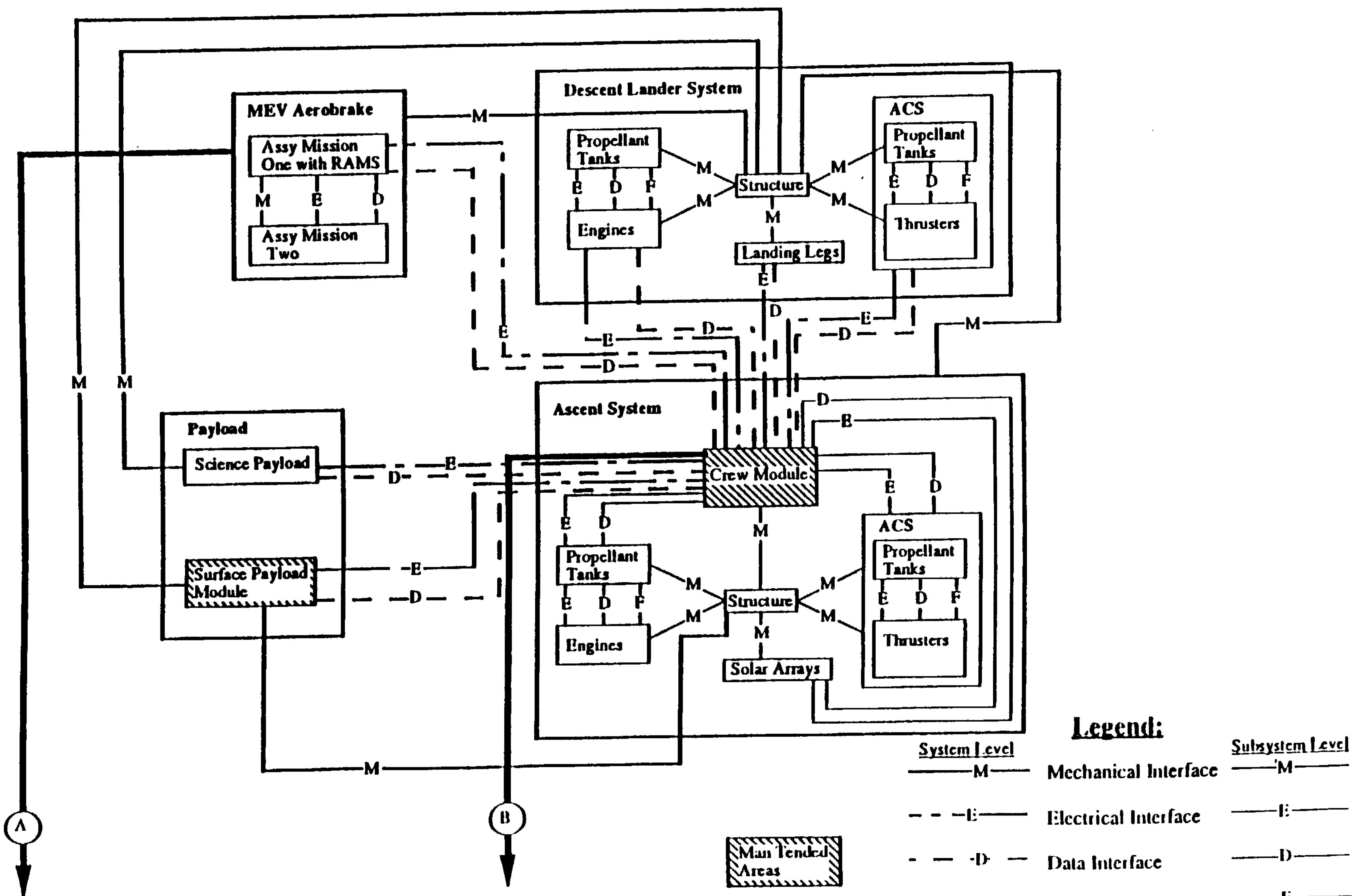
MEV System Interfaces



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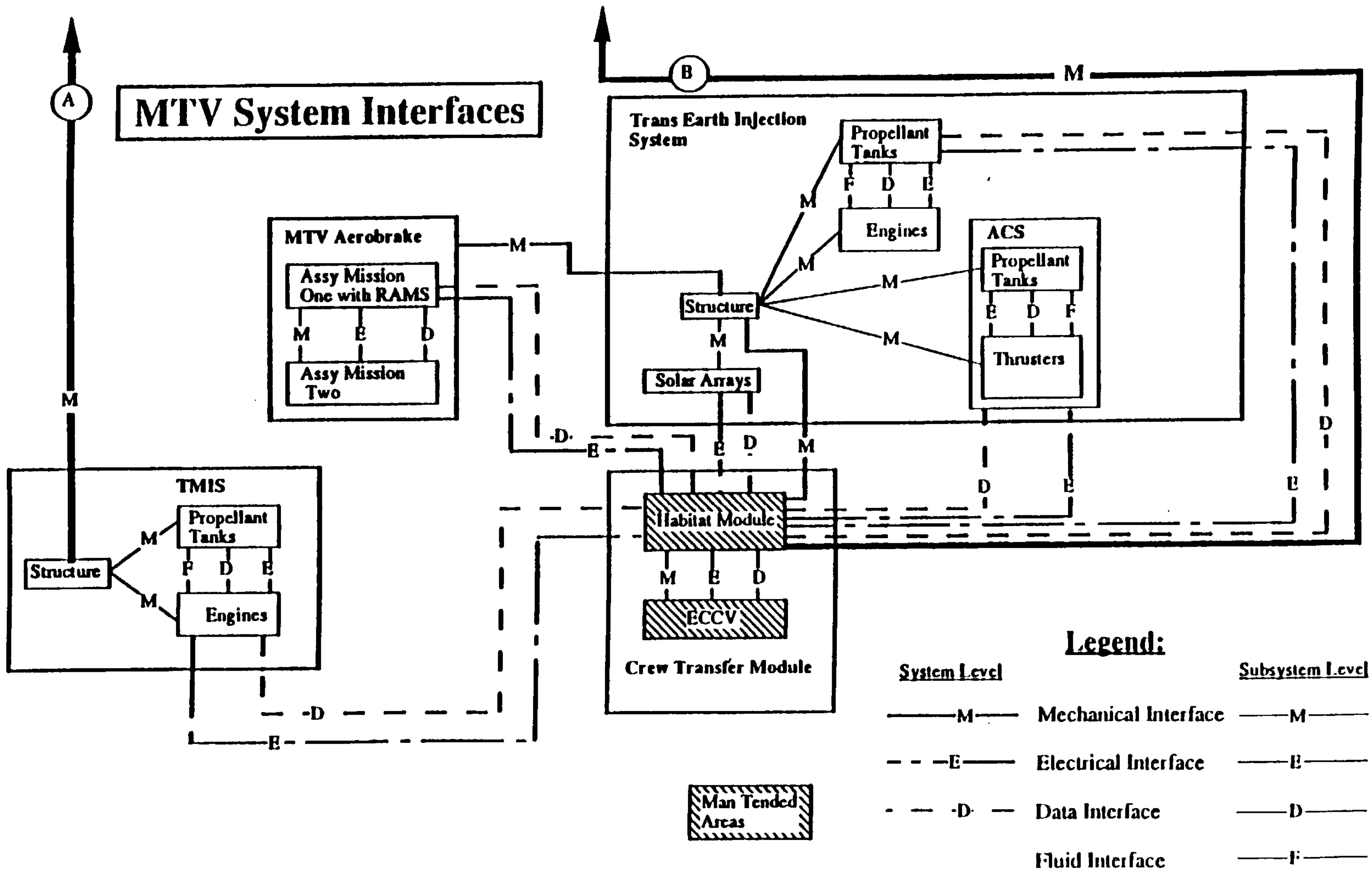
MMV System Interfaces

The following two charts show Cryo/Aerobrake System and Subsystem Interfaces. The interfaces shown are major interfaces, that is, one electrical interface may represent several electrical cables. The total number of component level interfaces has not been defined at this point.

/STCAEM/dks/12June90

MMV System Interfaces

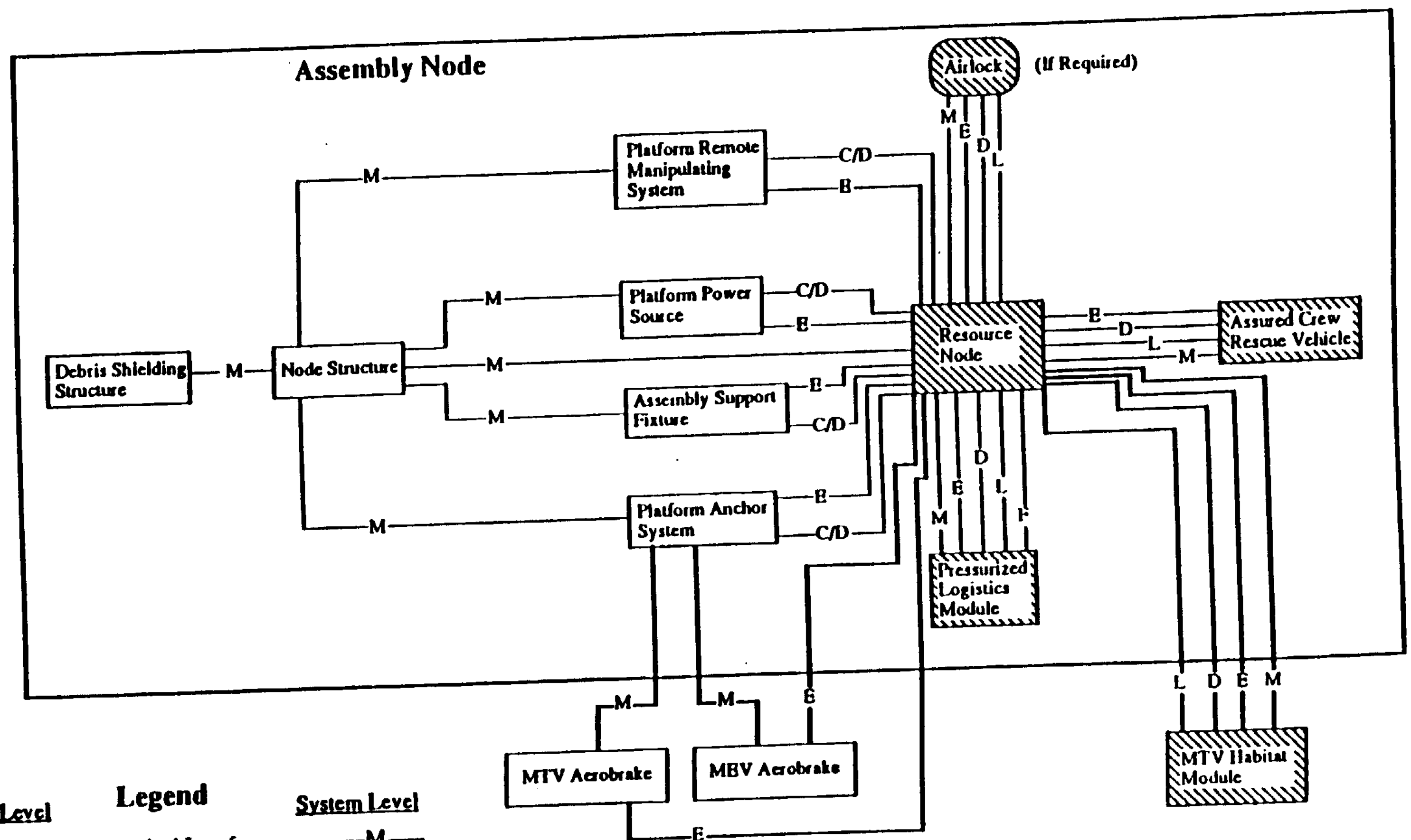


/STCAEM/dks/19March90

Assembly Node Interfaces

The following chart shows the Assembly Node interfaces to Cryo/Aerobrake. The Assembly Node requirements and equipment interfaces were developed by the On-Orbit assembly analysis.

Assembly Node Interfaces



Legend

Subsystem Level		System Level
—M—	Mechanical Interface	—M—
—D—	Data Interface	—D—
—E—	Electrical Interface	—E—
—F—	Fluid Interface	—F—
—L—	Life Support System Interface	—L—
—C/D—	Control/Data Interface	—C/D—

Man Tended Areas

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Sequential Interface Verification

The following chart defines the process of Sequential Interface Verification for Ground Processing of the Cryo/Aerobrake Vehicle.

Sequential Interface Verification

- Process of verifying the interfaces of the Mars Mission Vehicles elements without complete assembly.
- Elements are received and inspected at the assembly area.
- Internal test performed and certified by the contractor will not be repeated.
- Elements will be assembled to the level required to verify the interfaces from one element to another.
- Interfaces will be verified by flight hardware when feasible or by match mate devices/prototypes when necessary.
- Elements will be disassembled to payload configurations and processed for launch.

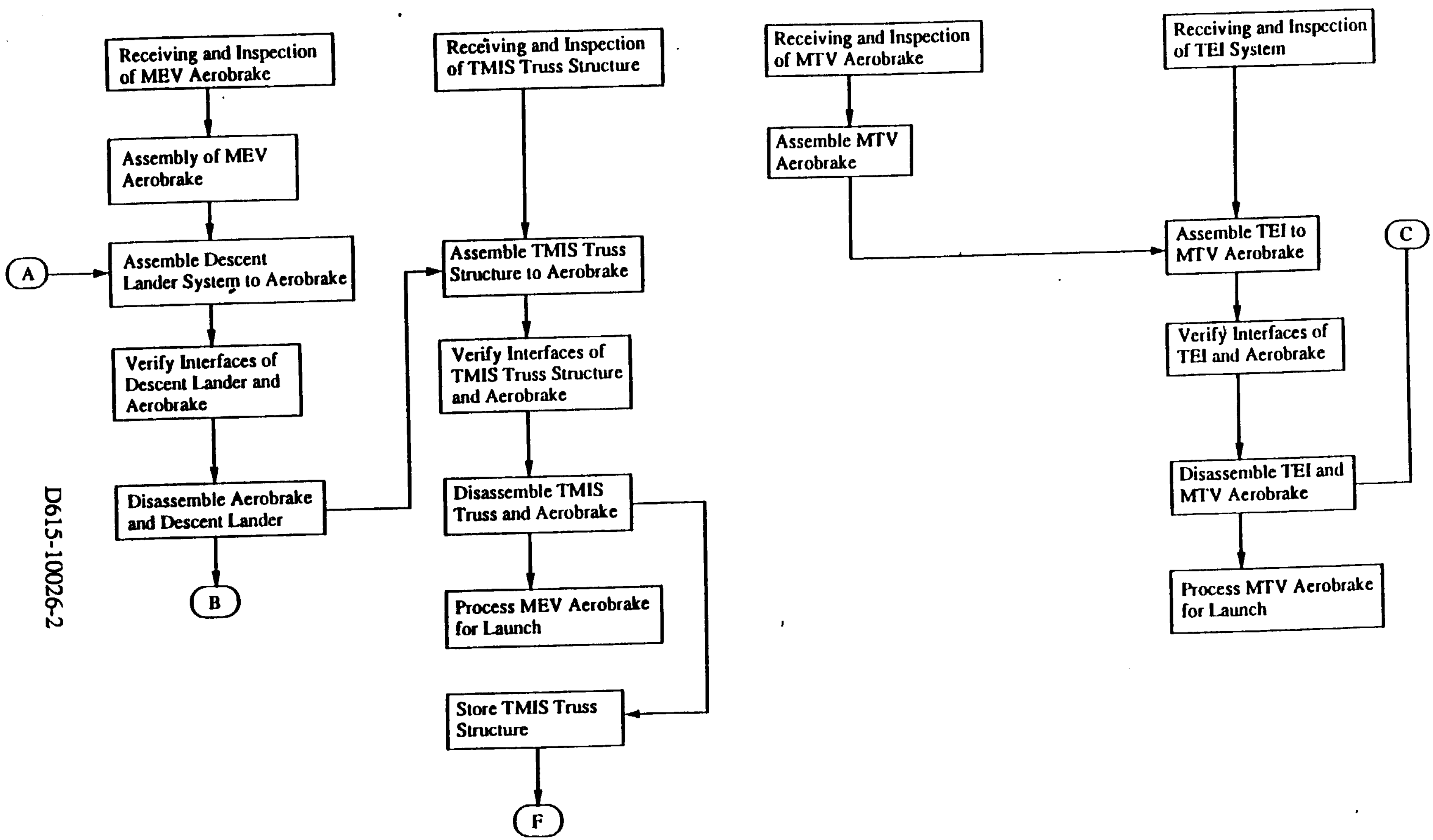
/STCAEM/dks/29May90

Ground Processing Functional Flow

The following three charts show the Functional Flow of Ground Processing for the Cryo/Aerobrake Vehicle. This Flow is a top level flow that shows the requirements for sequential interface verification.

/STCAEM/dks/12June90

Ground Processing Functional Flow



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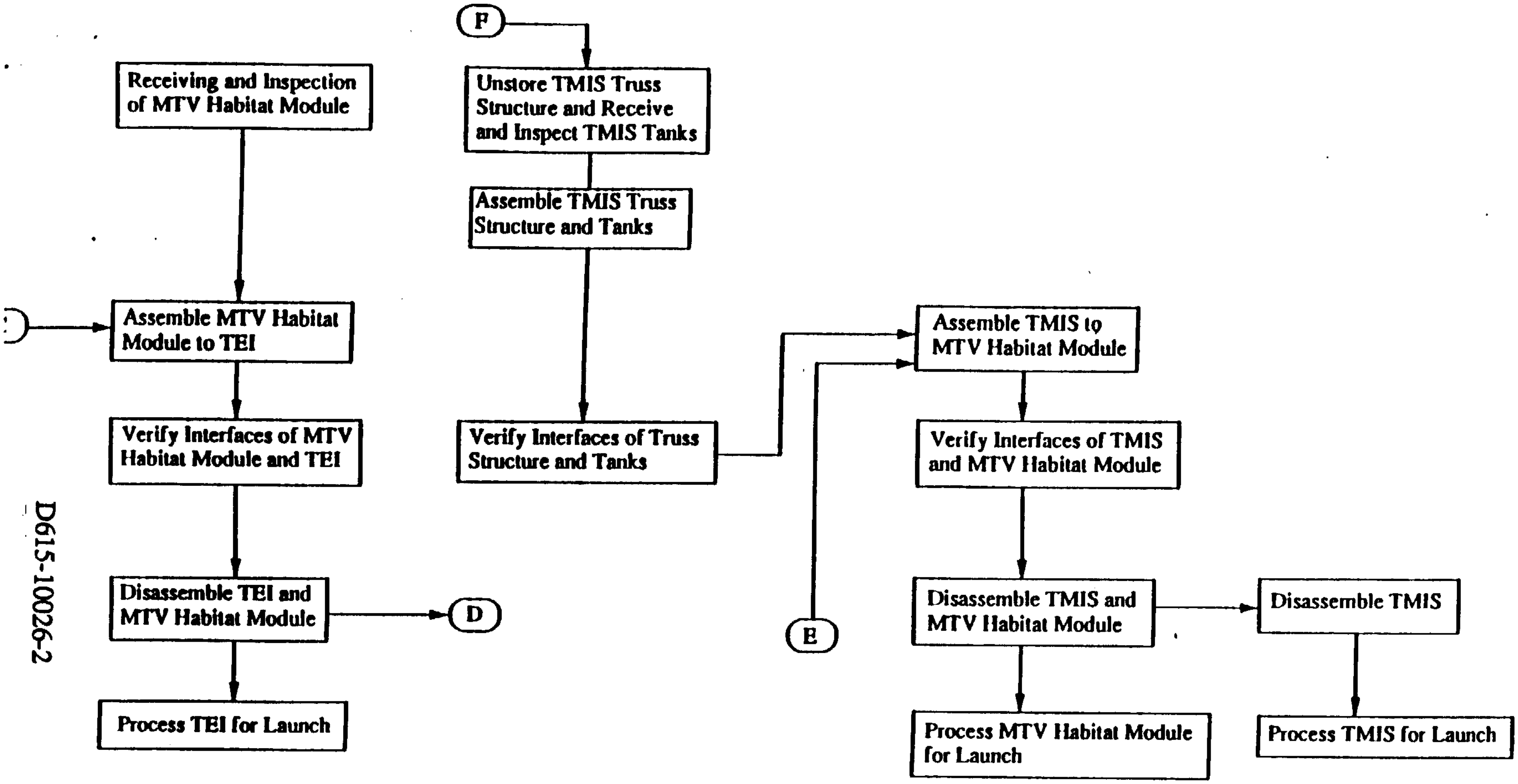
Ground Processing Functional Flow



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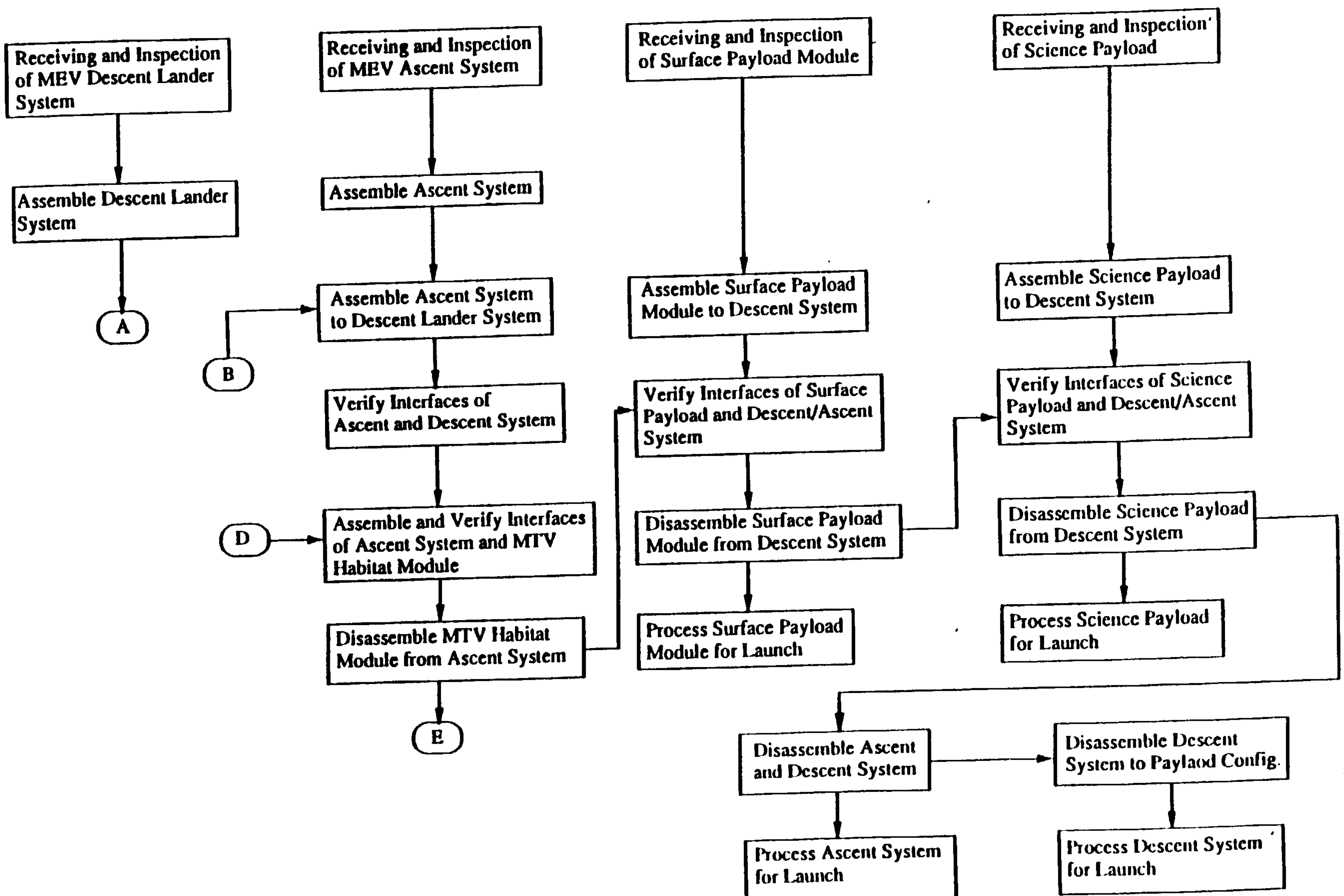
Ground Processing Functional Flow



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Cryo/Aerobrake Test Philosophy

The following chart is the summary of the Test Philosophy developed for this analysis. The complete Philosophy and Approach was included in the May Progress Report.

Test Philosophy

Purpopse: Establish criteria and overall test approach that verifies a system is flight ready and will accomplish its mission successfully

Goals:

- Reduce redundant testing
- Reduce man power requirements for ground processing
- Reduce overall cost
- Provide system operational history

Criteria:

- Self test software and peripheral equipment will perform mechanical, electrical and electronic system tests and readiness analysis in an autonomous fashion
- Redundant flight hardware (on-board systems) will be under continuous self check
- Physical interfaces will be self latching connectors
- Prototype systems will be utilized when feasible for ground processing activities
- Commonality of systems will be stressed

Ground Processing Facility Requirements

The following chart is a preliminary analysis of the Facility Requirements for Ground Processing of the Cryo/Aerobrake Vehicle.

.STCAEM/dks/12June90

Facility Requirements

- **Processing Facility Ground Rules:**
  - Utilize Standard Services: Cranes, Power, Communications, Clean Rooms, etc.
  - Make Unique Hardware Portable: Special Test Equipment, Work Stations, Handling Fixtures, etc.
  - Provide Large Volume Workspace that can be readily adapted to System Block changes,
  - Multiple Systems in Flow.
  - Provide for Hazardous System Processing

- **Equipment Requirements**
  - Overhead Crane
  - Flat Floor / Air Pallets
  - Standard Commerical Power
  - Uninterrupted Instrumentation Power
  - Environmental Control System: Humidity 50 +/- 5%   Temperature 75 +/- 5F
  - 100K Cleanliness Level
  - Closed Circuit Television
  - Facility GN2
  - Helium Supply
  - Shop Air
  - Fire Protection / Deluge
  - Shower / Eye Wash
  - Vacuum
  - Lightning Protection
  - Potable Water
  - Paging
  - Commerical Telephone
  - RF System
  - Operational Intercom System
  - Personnel Airlock
  - Grounding
  - Transportation/Ground Handling Fixtures

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VI. Implementation Plan

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Technology Needs and Advanced Plans

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Technology Issues - Cryogenic / Aerobraked Vehicle

I. Introduction

Technology issues relating to the reference vehicle are presented in this section. Some of the charts are also included in the NTR, NEP, and SEP IP&ED documents. The focus of this section will be to bring out those issues important to the reference cryogenic vehicle from these charts, and to present a series of technology level requirements necessary for the reference vehicle.The most important technology development needs for this option are in the areas of high energy aerobraking, and cryogenic fluid storage and management

II. Technology commonality Issues

The following nine charts lay out the important technology commonality issues between the major propulsion options as well as across the seven major mission architectures identified in this study. The reference vehicle exhibits commonality, and therefore is a good "building block" for the other vehicles in several important areas. The transfer crew module is substantially the same as for all the other options. The MEV is identical across all vehicle options, except for the cryogenic propellant management and storage issues. The demands placed on the avionics system for the chemical system are similar to those for the NTR, and probably greater than those needed for the low thrust NEP and SEP options. Finally, in-space assembly issues should be similar for the reference and NTR vehicle, with the exception of the related nuclear issues associated with the NTR. Assembly issues relating to the NEP and SEP, while duplicate in some areas, will be unique in most areas.

The seven identified Lunar/Mars mission architectures verses the required component technologies, enabling and enhancing, are shown on the next set of charts and facing page text. Many of these component technology issues are common across the listed architectures. These issues are for the entire integrated architectures, and do not necessarily refer specifically to the reference vehicle. Cryogenic/aerobraked vehicles are used in most of the architectures for initial Mars missions, and for all early Lunar missions. The areas of high thrust cryogenic propulsion, and high energy aerobraking are the primary areas of technology development concern for the reference option.

III. Technology Development Concerns

As noted before, many of the identified critical and high leverage technology development issues are common across all four major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique cryo/AB technology issues include high energy aerobraking, and large advanced space engine advanced development. Enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

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IV. Cryo/Aerobraked Vehicle Technology Requirements

Technology performance levels required for the reference cryogenic vehicle are outlined in the next eight charts. These are not intended to be the levels needed for a minimum vehicle, but serve mainly to document the levels required to accomplish the identified reference mission profile with the vehicle model as configured. Changes to these specifications would not necessarily affect the feasibility of a chemical Mars mission, but would change the reference vehicle configuration. The list also includes operational requirements which could drive technology development or advanced development. An example of this could be the requirement of wet launched tanks, rather than filling on orbit, which would affect tank design, and possibly in-space thermal performance.

V. Cryo/Aerobraked Technology Development Schedule

The final chart in this section is a proposed technology development schedule for the nuclear electric propulsion option. The schedule shows that, given a FY '91 start, the SEP vehicle could be ready for a Mars mission in the 2009 timeframe. A full scale decision point is also highlighted during year 7. This is the point where a commitment should be made for full scale funding and development of the program.

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Technology Commonality and Differences

System/Subsystem	Reference (Cryo-A/B)	NTR Vehicle	NEP Vehicle	SEP Vehicle
Crew Systems/Habitats Life Support, rad. prot., hab. struct., & airlock/EVA	Long duration life support system derived from SSF proven system. LTV crew module evolves to MTV; common LEV/MEV habitat system. Mars surface habitat derived from proven Lunar design. Mars surface TCS requires additional technology advances to deal with unique heat rejection problems. All extended missions (>2-3 d) require solar flare radiation protection. Hab systems common across mission architecture. Shorter mission LSS sized for free return abort contingency. Minimum mass airlock could be shuttle-evolved.			
Power System & Thermal Control	Deployed solar array system; low power (~50-75 kW). Low temp heat rejection (~400°K)	Common to reference vehicle system	Nuc. /Rankine or Brayton cycle energy conv. sys. Very high power level (up to 200 MW). High temp heat rejection (~1000°K-main cycle).	Solar-electric energy conversion. High power (~10 MW or greater) level. Moderate temperature radiators (400 - 650 K).
Propellant Management & Storage	Long term storage of H_2 & O_2 for Earth & Mars orbit, and deep space environ. necessary with minimal boiloff. Low-g fluid gaging, acquisition, and transfer highly enhancing or enabling for all missions. NTR requires common techniques for LH_2 fuel.		Argon propellant management system can be similar to LOX storage system, but without the safety constraints associated with an oxidizer.	
Propulsion System	Advanced cryogenic space engines with >475 sec Isp, and ~30 klb to ~200 klb thrust.	NERVA derived /advanced NTR system with higher Isp (up to 1050 sec vs. 850 sec.)	Rankine or Brayton cycle conversion system driving cluster of Ion thrusters for NEP. Same thrusters for SEP. Number of thrusters depends on available thruster size and required redundancy.	
Aerobraking Lunar	Low L/D - AFE derived for Earth capture.	Not needed for Lunar NTR (propulsive capture@ Earth)	Not needed for NEP or SEP.	
Mars	Higher L/D necessary - structure and TPS technology base.	Only low energy lander aerobrake needed, since entire vehicle, including MEV is propulsively captured at Mars. Can be common with earlier cryo A/B vehicle, unless crossrange constraints require higher L/D design.		
Avionics	Avionics system hardware may be common for Lunar or Mars (or L/M growth)		Avionics system required for low & continuous thrust vehicles are lower than for Cryo A/B or NTR vehicle.	
Assembly & Checkout	Common assembly facility & equip. for most mission vehicles. Assembly time in LEO, and thus M/D protection level is varied. Mars vehicle requires launch & assembly of large (~ 30 m vs. 20 m for Lunar) aeroshell. Nuclear vehicles (NTR & NEP) may face political constraints on launch & assembly of vehicle. Assembly & operation may be necessary from nuclear safe orbit.			Severe LEO debris environ. damaging to solar arrays. Spare set of arrays may be necessary. MEV A/B launch & assembly needed.

Required Technologies vs. Alternative Mission Architecture

A set of required technologies for the seven identified alternative mission architectures outlined in the evolotionary concepts section is presented. The purpose of this matrix is to provide a preliminary comparison of technology development needs for the alternative architectures. The matrix also serves to better define the architectures. From this top level matrix, a more detailed set of technology requirements can be derived. A set of accommodating technologies can be compiled for needs areas where options exist. Finally, the technology areas can be prioritized as enabling and enhancing, and a return on investment performed for identified high leverage technologies. This portion of the matrix includes most of the cryogenic management issues. Enabling technologies are represented by the filled circle, and enhancing technologies by the open circle. Extensive low - g cryogenic propellant launch, acquisition and transfer refers to the Mars conjunction case, and the mass driver option, where propellant will be used for the transfer vehicles, which will be parked in a low - g environment (Lunar or Mars orbit, or libration staging point). The Mars cycler orbit case includes a question mark for the long term cryogenic storage system, because the necessary thrust levels and type of propulsion system are undetermined at this time.

Required Technologies vs. Alternative Mission Architecture

	Low boiloff cryogenic propellant storage system (1-3 yr)	Low boiloff cryogenic propellant storage system (15 - 60 d)	Low - g fluid acquisition and transfer	Extensive low - g cryogenic propellant launch, acquisition, and transfer	Cryogenic tank integrity monitor	Cryo fluid rematible umbilical	Lunar LOX production, liquification, and transfer technology	Mars O2 production, liquification, and transfer technology
Mars NEP Alternative Architecture	○	●	●		●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●		●	●	●	
Mars SEP Alternative Architecture	○	●	●		●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●	●	
Mars Cycler Orbits Alternative Architecture	?	●	●		●	●	●	
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●	●	● + H2
Lunar / Mars NEP Alternative Architecture	○	●	●		●	●	●	●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix section represents the major aerobraking concerns. The aerobraking energy columns for Mars and Earth capture digresses from the format in order to illustrate the energy levels, and therefore, the level of technology development needed for the various architectures. Aeroheating predictions, reusable aerobrake TPS, advanced GN&C, and TT&C follow along with the high and medium energy missions. Again, a question mark is shown for the Mars cycler orbit case. Reusable TPS for Earth return cannot be determined as a technology development concern until the aeroheating load at Mars can be determined for the cycler orbits. Further mission design efforts must be carried out before an estimate on this can be made.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Earth return aerobrake energy	Mars capture aerobrake energy	Mars lander aerobrake	High performance aerobrake structure	Aerobrake assembly and test	Aeroheating prediction (Earth and/or Mars)	Reusable aerobrake TPS for Earth return	GN & C to protect TPS	Advanced high accuracy and rate TT & C	In space AR&D / assembly
Mars NEP Alternative Architecture	Low	Low	●	●	●					●
Lunar/Mars NTR Alternative Architecture			●	●	●					●
Mars SEP Alternative Architecture	Low	Low	●	●	●					●
L2 Node / Mass Driver Alternative Architecture	High	High	●	●	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	High	High	●	●	●	●	?	●	●	●
Mars Conjunction/Direct Alternative Architecture	Medium	Medium	●	●	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	Low	Low	●	●	●					●

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

This matrix area represents the major propulsion issues, with the exception of the radiation protection system, for the baseline and alternative mission architectures. This system, which uses the inert and can waste for radiation shielding, can be enhancing, while a GCR and ALSPE shelter is enabling for all mission architectures. Again, due to the undefined MARS cycler orbit trajectories, it is questionable as to the need for a large cryogenic space engine. A H_2-O_2 ACS/RCS system is noted as enabling for each option, as it will be for any option over a baseline storable system. A lunar orbital momentum storage and transfer device such as a bolo can be enhancing for all missions, after an initial launch and assembly penalty for the massive (~1000 t) device.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Large (150 - 200 klb) cryogenic advanced space engine	Small (15 - 30 klb) cryogenic advanced space engine	H2 - O2 ACS/RCS	Multi - MW space based nuclear electric power	Multi - MW space based nuclear thermal power	Surface nuclear electric power	Multi MW solar power system (arrays and handling equip.)	Radiation protection (system to inert & can waste)	Mass driver / rail gun technology	Lunar orbital momentum transfer device (Bolo)
Mars NEP Alternative Architecture		●	○	●		●		●		○
Lunar/Mars NTR Alternative Architecture		●	○		●	●		●		○
Mars SEP Alternative Architecture		●	○				●	●		○
L2 Node / Mass Driver Alternative Architecture		●	○			●		●	●	○
Mars Cycler Orbits Alternative Architecture	?	●	○			●	●	●		○
Mars Conjunction/Direct Alternative Architecture	●	●	○			●		●		○
Lunar / Mars NEP Alternative Architecture		●	○	●		●		●		○

● - Enabling

○ - Enhancing

Required Technologies vs. Alternative Mission Architecture (Cont.)

The final section of the matrix is not as illustrative as the others, in that all of the listed technologies are enabling, with the exception of a closed ecological life support system, which is significantly enhancing for all identified mission architectures.

Required Technologies vs. Alternative Mission Architecture (Cont.)

	Autonomous health monitoring and check-out	High data rate comm. or high performance compression	DMS/system diagnostics. Art. intell/neural nets/high processing rate GN&C	Long duration refurbishable crew habitat	Long duration BCLSS	CELSS
Mars NEP Alternative Architecture	●	●	●	●	●	●
Lunar/Mars NTR Alternative Architecture	●	●	●	●	●	●
Mars SEP Alternative Architecture	●	●	●	●	●	●
L2 Node / Mass Driver Alternative Architecture	●	●	●	●	●	●
Mars Cycler Orbits Alternative Architecture	●	●	●	●	●	●
Mars Conjunction/Direct Alternative Architecture	●	●	●	●	●	●
Lunar / Mars NEP Alternative Architecture	●	●	●	●	●	●

Mars Reference Vehicle Technology Requirements

STCAEM/jrm/6F

I. TMIS

A. Cryogenic storage system

1. Thermal protection system - MLI over foam. (1" foam; ~ 1" MLI)
2. Tanks launched wet.
3. Thermodynamic vent coupled to a single vapor cooled shield.
4. Topoff before Earth departure.
5. ~ 6 months in LEO before use.
6. Negligible boiloff loss after topoff.

B. Propulsion

1. Isp = 475 s
2. Thrust = 150 klb/engine
3. Advanced space engine.
4. Nozzle area ratio = 400
5. No throttling requirements.
6. Gimbal angle (nominal) = 10°
7. Up to 3 burns for departure maneuver (2 restarts).
8. Engine out capability (crossfeed propellant lines).
9. No specified engine cycle.
10. In-space changeout capability.
11. Off vehicle preflight checks.
12. No retraction / extension required.

Mars Reference Vehicle Technology Requirements (cont.)

C. Structure
1. Material - metal matrix composites, advanced alloys, and organic matrix composites.
2. Meteor/debris protection provided for tanks and plumbing.

D. Avionics
Piggybacked on MTV.

E. Power
1. Level : < 1 kW
2. System: Auxiliary power units on engine pod; piggybacked on MTV for back-up.

F. Assembly
1. Off station assembly.
2. Degree of assembly: Separate tanksets / propulsion modules connected in LEO to form propulsion stage.

II. MTV

A. Cryogenic storage system
1. Thermal protection system - MLI; 100 layers on H_2 & O_2 tanks (2").
2. Tanks launched wet - no transfer other than topoff before Earth departure.
3. Thermodynamic vent coupled to a series of vapor cooled shields on the H_2 tank, and one on the O_2 tank.
4. Topoff in LEO before Earth departure.
5. ~9 months in LEO before Earth departure.
6. Boiloff loss of < 10% before Mars departure.

Mars Reference Vehicle Technology Requirements (cont.)

STCAEM/jrm/6F

B. Propulsion

1. Isp = 475 s.
2. Thrust = 30 klb/engine.
3. Nozzle area ratio = 400.
4. No throttling requirements.
5. Gimbal angle (nominal) = 10°
6. M/D shield for plumbing & tanks.
7. 3 burns @ 4 - 6 month intervals - minimal degradation.
8. 2 restart capability.
9. Engine out capability (crossfeed propellant lines).
10. Expander cycle.
11. In-space change out capability.
12. Off vehicle preflight checks.
13. No retraction/extension required.

C. Structure

1. Vehicle
   a. Metal matrix composites / advanced alloys / organic matrix composites.
   b. Micrometeoroid protection for habitat structure (shell and insulation).

Mars Reference Vehicle Technology Requirements (cont.)

2. Aerobrake
   a. L/D = 0.5
   b. Crossrange: NA
   c. Vhp = 7.07 km/s.
   d. Max-g loading = 6.
   e. Max. temperature = 4000° F.
   f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   g. TPS material: Advanced reradiative tiles.
   h. Relative wind angle (reference) = 20°.

D. <u>Avionics</u>
1. Planetary vicinity -
   a. Relative velocity error = 100 m/s.
   b. Relative position error = 25 km.

2. System -
   a. Relative velocity error = 100 m/s.
   b. Relative angle error = 0.5°.

E. <u>Power</u>
1. Level - 15 kW.
2. System: Solar arrays with battery storage (NiCad).
3. Back up system: NA

Mars Reference Vehicle Technology Requirements (cont.)

STCAEM/jrm/6F

F. <u>Assembly</u>

1. Off station assembly.
2. Assembly level (complexity): TBD

G. <u>Habitat</u>

1. ECLSS: Space Station Freedom derived system with similar degree of closure; potable H_2O from cabin condensate; CO_2 reduction/regeneration; Hygiene H_2O from urine processing. CELSS to be evaluated.

2. Structure
   a. 2219 - T8 aluminum pressure vessel.
   b. Pressurized to 20 psig on launch for structural integrity.
   c. Insulation & M/D shield external to pressure shell.
   d. No penetrations in end domes.
   e. Radiation storm shelter provided, and configured to utilize equipment & supplies as partial shielding.
   f. External space radiator integral with M/D shield.

3. Cabin repressurizations: 2+ (outbound emergency could use propellant for repress.)
4. Spares: 15% of active equipment - component level.
5. Redundancy: Two complete and separate systems for life critical systems + spares. Component changeout capability.
6. Residence time = 535 days.
7. Science: Transit science as allowed by individual mission.
8. EVA capability: EVA suits provided for all crew; EVA waste fluid recovery for ECLSS.

Mars Reference Vehicle Technology Requirements (cont.)

H. ECCV

1. Apollo size & style as a starting point
2. Open ECLSS (LiOH, no H_2O recovery).
3. Residence time: 2 - 3 days.
4. Propulsion: RCS only.

III. MEV

A. Cryogenic storage system

1. Thermal protection system: 100 layers of MLI for H2 and O2 tanks (2").
2. Tanks: double wall tanks with vacuum annulus;
   low thermal conductivity support system for inner tank.
3. Thermodynamic vent: Simple design for gravity field.
4. Tanks launched dry and filled prior to descent, from MTV tanks, or refrigerated. (no boiloff prior to descent)
5. Stay time from 30 - 600 days on Mars surface.
6. Boiloff level < 20% for surface stay.

B. Propulsion

1. Isp = 460 sec.
2. Thrust = 30 klb / engine.
3. Nozzle area ratio = 200.
4. Throttleability = 15:1.

Mars Reference Vehicle Technology Requirements (cont.)

STCAEM/jrm/6F

B. <u>Propulsion</u> (cont.)
- 6. Gimbal angle (nominal) = 10°.
- 7. No restart capability necessary for nominal case.
- 8. Space storage time between burns : NA.
- 9. Engine out capability (crossfeed propellant lines).
- 10. Expander cycle.
- 11. In-space changeout capability.
- 12. Off vehicle preflight checks.
- 13. Retraction / extension capability.

C. <u>Structure</u>
1. Vehicle
   - a. metal matrix composites / advanced alloys / organic matrix composites.
   - b. Micrometeoroid protection for tanks and plumbing.

2. Aerobrake
   - a. L/D = 0.5 to 1.0
   - b. Crossrange: 1000 km.
   - c. Vhp = 7.07 km/sec.
   - d. Maximum g loading: 6.
   - e. Maximum temp: TBD (estimated 3100° F).
   - f. Structure material: Carbon Magnesium ribs (σ_{ult} = 200 ksi) bonded to titanium honeycomb shell.
   - g. TPS material: Advanced reradiative tiles.
   - h. Relative wind angle (reference) = 20°.

Mars Reference Vehicle Technology Requirements (cont.)

D. Avionics
1. Error without beacon = 1 km.
2. Touchdown error = 1 m/s.
3. Obstacle avoidance capability.

E. Power
1. Level: ~ 2.5 kW.
2. System: fuel cells (regenerable).
3. Back-up system: abort to orbit.

F. Assembly
1. Off station assembly.
2. Assembly level (complexity): TBD

G. Habitat
1. ECLSS: open system; stored potable H_2O; LiOH CO_2 adsorption.
2. Structure
   a. Aluminum (2219 - T8) pressure vessel.
   b. Overpressurized on launch for structural integrity.
   c. Insulation and micrometeoroid protection external to pressure vessel.
   d. No penetrations in end domes.
   e. No radiation shelter provided in MEV.
   f. External space radiator integral with micrometeoroid shield.
3. Repressurizations: 2.
4. Spares: 15% of active equipment mass; component level.
5. Redundancy: EVA suits as backup to cabin repressurization.; no system level ECLSS redundancy required due to low complexity open system.
6. Residence time: ~3 days (surface systems support surface stay).
7. Science: none.
8. EVA capability: provided for all crew; transferred from MTV.

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Critical Lunar/Mars Reference Technology Development Concerns

A preliminary set of critical technology development concerns was constructed for the Lunar/Mars reference missions. Its purpose is to show a top level representation of the areas which could prove enabling for the reference Lunar and/or Mars missions, without further concentrated research and development, flight testing, and/or precursor missions. Aerobraking may prove enabling for most Lunar and Mars missions, and significantly enhancing for the rest, primarily due to reduced demands on limited Earth to orbit launch capability and lower launch costs. Aeroheating prediction codes cannot be validated without further experimental data (flight or ground simulation data). The degree of development needed for aerobrake TPS materials will be determined by these predictions. Low gravity human factors, to be evaluated on SSF, may affect vehicle design significantly. Foe example, vehicle designs must accommodate artificial - gravity until a need level can be determined from space station based research. Finally, precise mission design, incorperating advanced tracking, telemetry, and GN&C must be verified to accommodate aerobraking and automated rendezvous & docking requirements.

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

A preliminary set of high leverage technologies was assembled for the Lunar/Mars reference missions. These technologies are enhancing for most, and in some cases, all identified mission architectures. Aerobraking will be significantly enhancing for all Lunar and Mars missions where it is not identified as enabling. Other aerobraking issues which could prove enhancing are lightweight reradiative or ablative TPS material, and ECCV vs. aerocapture of MTV at Earth. Low - g propellant handling and low boiloff cryogenic storage are also very enhancing for any missions where it is not enabling. Advanced propulsion options such as NTR, GCR, SEP, and NEP may prove to be high leverage technology options to baseline cryogenic propulsion systems. Finally, developments in advanced materials can be significantly enhancing in a variety of areas.

Preliminary Identified Lunar/Mars Reference High Leverage Technology Issues

Technology	Comments
Aerobraking - Mars Capture (vs. propulsive cap.)	- Aerocapture at Mars can reduce IMLEO >50% over propulsive capture
Aerobraking - Earth Capture (vs. ECCV)	- ECCV reduces IMLEO and thermal protection system (TPS) requirements. - Reusable MTV can reduce life cycle cost.
Aeroshell TPS (reradiative vs. ablative)	- Reusable aeroshell requires rerad. TPS at Mars (or thick lightweight ablator), and ablative at Earth. - Further materials and processes advances or low energy mission may allow Earth/Mars reradiative TPS.
Advanced Long Term Cryogenic Storage Technology	- Cryogenic boiloff reduction technologies such as advanced MLI design and application, VCS, para to ortho H2 conv., and thermal disconnect struts, can reduce IMLEO significantly with low R & D effort - Longer missions offer greater IMLEO savings potential
Low - g Propellant Transfer	- Low - g propellant transfer technology enhancing for all Lunar/Mars mission arch., and enabling for some Lunar missions.
Efficient Cryogenic Refrigeration System	- Cryogenic refrig system can reduce vehicle mass and enhance system reliability at the expense of an increased power level.
O2 - H2 ACS / RCS	- O2 - H2 ACS/RCS (Isp = 400 s) reduces system mass over lower Isp storables
High Isp Advanced Space Engine	- High Isp advanced space engine (Isp = 485 s) enhances all mission phases for all mission arch.
NTR Propulsion System	- NTR propulsion system for the TMI, Lunar transfer, and Mars transfer stages
Advanced In - Space Assembly Techniques	- Launch vehicle capability drives on - orbit assembly level. - Degree of on - orbit assembly capability affects vehicle configuration, ground assembly/processing, and launch manifesting.
Advanced Materials Development	- Advanced materials such as metal and organic matrix composites reduce system inert mass, strength, and/or manufacturing costs. - Some advanced M&P may prove enabling for some mission arch. (ex:Mars/ Earth capture aerobrake)

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Schedules

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Technology Development Concerns and Schedules - Cryogenic All Propulsive Vehicle

Critical technology development issues relating to the reference CAP vehicle are presented in this section. Where applicable, the same charts are also included in the CAB, NTR, NEP, and SEP IP&ED documents. The focus of this section will be to bring out the most important issues relating to the reference cryogenic all propulsive vehicle, and to present preliminary technology development schedules for these issues. The issues are presented here in outline form, beginning with the most important, with accompanying schedules wherever possible.

Cryogenic Propulsion and Fluid Management

With the absense of high energy aerobraking for the all propulsive mission, cryogenic propulsion and fluid management becomes the most important technology development concern in the area of vehicle benefits. The high Isp of a LH2-LOX system (460-480 s) may prove enabling for an all propulsive mission due to the massive vehicle sizes which could result from the lower Isp (280-360 s with metallic gels) storable systems. The long term storage and low-g fluid management of cryogenic fluids, along with long lifetime, in-space restartable cryogenic engines are the major technology development concerns for a cryogenically fueled vehicle. Preliminary technology schedules are presented for space based cryogenic engines, and cryogenic fluid system development for both Lunar and Mars applications. The cryogenic space based engine development effort begins with the planned AETB work at LeRC, and continues on to development work for a large engine for Mars applications. The cryogenic fluid systems schedule includes Earth-based thermal control and selected management (tank pressure control, liquid acquisition device effectiveness, etc.) tests, as well as planned flight experiments to carry out system and subsystem validation tests.

Vehicle Avionics and Software

Although the technology readiness level of vehicle avionics and software is ahead of many of the other technology areas listed in some respects, the demands on the system in the areas of processing rate, accuracy, autonomous operation, and status/health monitoring will drive technology and advanced development in areas not fully defined at this point. Software requirements cannot be fully determined until the vehicle design is at a more finished stage than the current levels. A preliminary schedule for autonomous systems development is presented. The decision points for full scale development The communications system options can be more fully defined before a final vehicle design is produced, however. A technology development schedule for advanced communications is presented.

Life Support

A reliable, redundant long term life support system will be enabling for future exploration missions. The degree of closure of, and the reliability of the system are the major technology development concerns. Low-g human factors determination will also be an important technology consideration which will drive vehicle design. An integrated schedule of the major areas of the life support technology development task are presented. It includes radiation shielding and materials, regenerative life support, and EVA systems development. As before, the points where Lunar and Mars full scale development decisions can logically be made in the technology program are highlighted.

Aerobraking (low energy)

Low energy aerobraking will offer mission benefits in the areas of decreased demands on the descent propulsion system, and improved crossrange capability. This area presents a variety of issues for technology development including high strength to mass ratio structural materials, high temperature thermal protection systems (although not as high as for high energy aerobraking), avionics, assembly and operations, hypersonic test facilities and computer codes, and Mars atmosphere prediction. High strength structural material options include metal matrix composite, organic matrix composite, and advanced carbon-carbon elements. Other structural considerations include load distribution and attachment of payload for aerocapture, and ETO launch and assembly of large structures. Thermal protection systems issues include low mass ablative and reradiative materials, and structure/TPS integration issues. The aerobrake maneuver will place considerable demands on the vehicle avionics system with the need for real time trajectory analysis, and vehicle guidance and control. The launch and assembly of the large aerobrake structure will present ground and space assembly and ops problems which will require technology and advanced development in both the areas of design and operations. Finally, computational analysis and atmosphere prediction capability will be critical in the development of a man-rated aerobrake for Mars use. A preliminary development schedule for Lunar and Mars aerobrake technology development is presented. It includes the major milestones for both ground and flight testing. The points where a Lunar and Mars full scale development decision can be made are also highlighted on the schedule. It should be noted that this schedule was built with high energy aerobraking in mind, and will possibly be compressed to some degree if only low energy aerobraking is developed.

In-Space Assembly and Processing

The in-space assembly and processing of large space transfer vehicles will present a variety of technology advanced development challenges, particularly for the large LTV and MEV aerobrakes. As shown on the accompanying schedule, extensive ground tests must occur before any orbital work can be initiated. The vehicle designs will be driven to a large degree by the assembly facilities and technologies seen as being available during the vehicle buildup sequence.

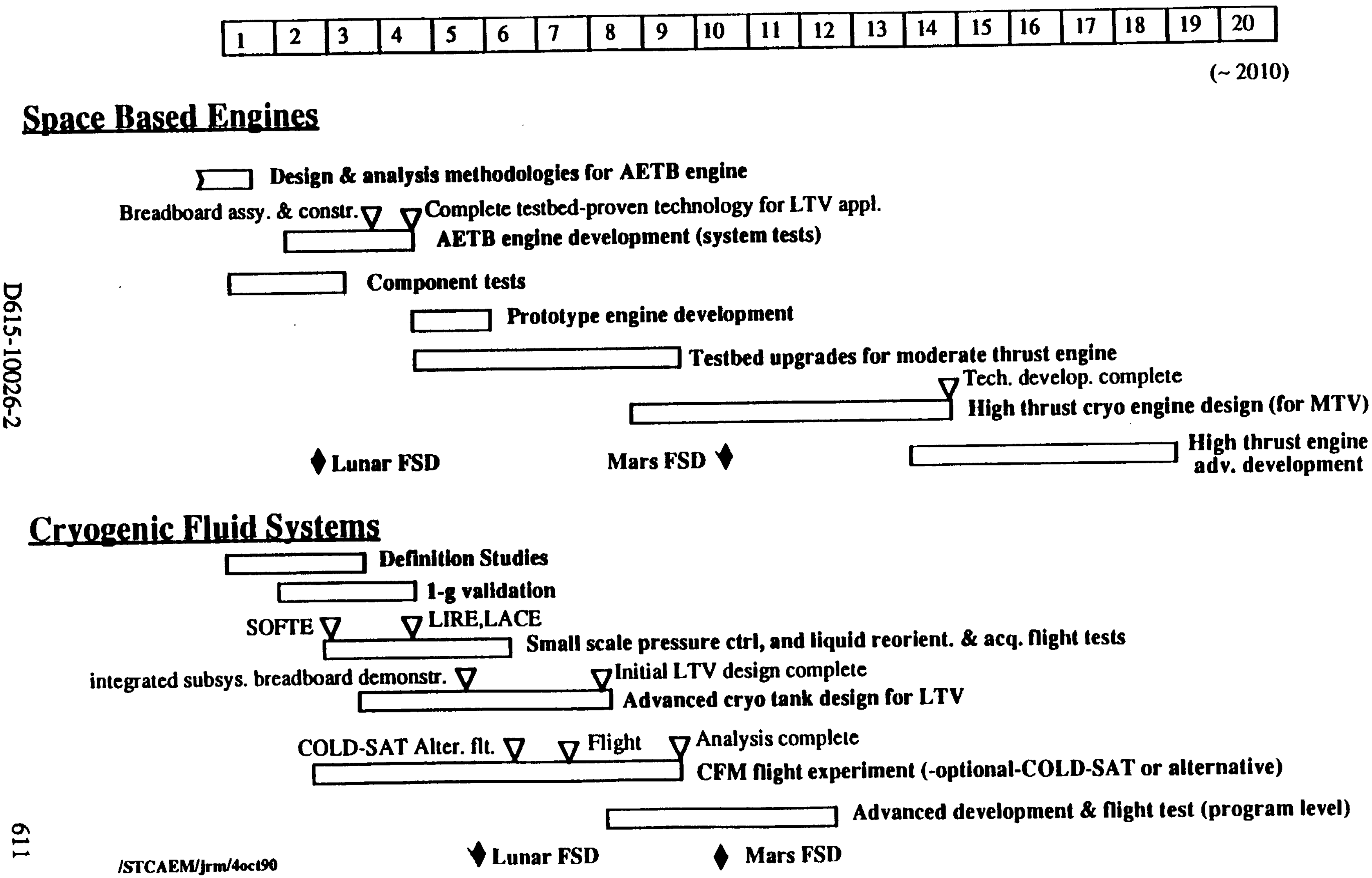
Summary

As noted before, many of the identified critical and high leverage technology development issues are common across all of the major vehicle options. Common critical technology issues include low-g human factors, autonomous system health monitoring, long term cryogenic storage and management (H_2, and possibly O_2 for ECLSS), long duration ECLSS, radiation shelter material and configuration, and in-space assembly. Unique cryo all propulsive technology issues center around large advanced space engine advanced development. Enhancing technologies include cryogenic refrigeration (lander tanks), O_2-H_2 RCS, advanced in-space assembly techniques, higher Isp cryogenic engines, and advanced structural materials development.

ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules

BOEING



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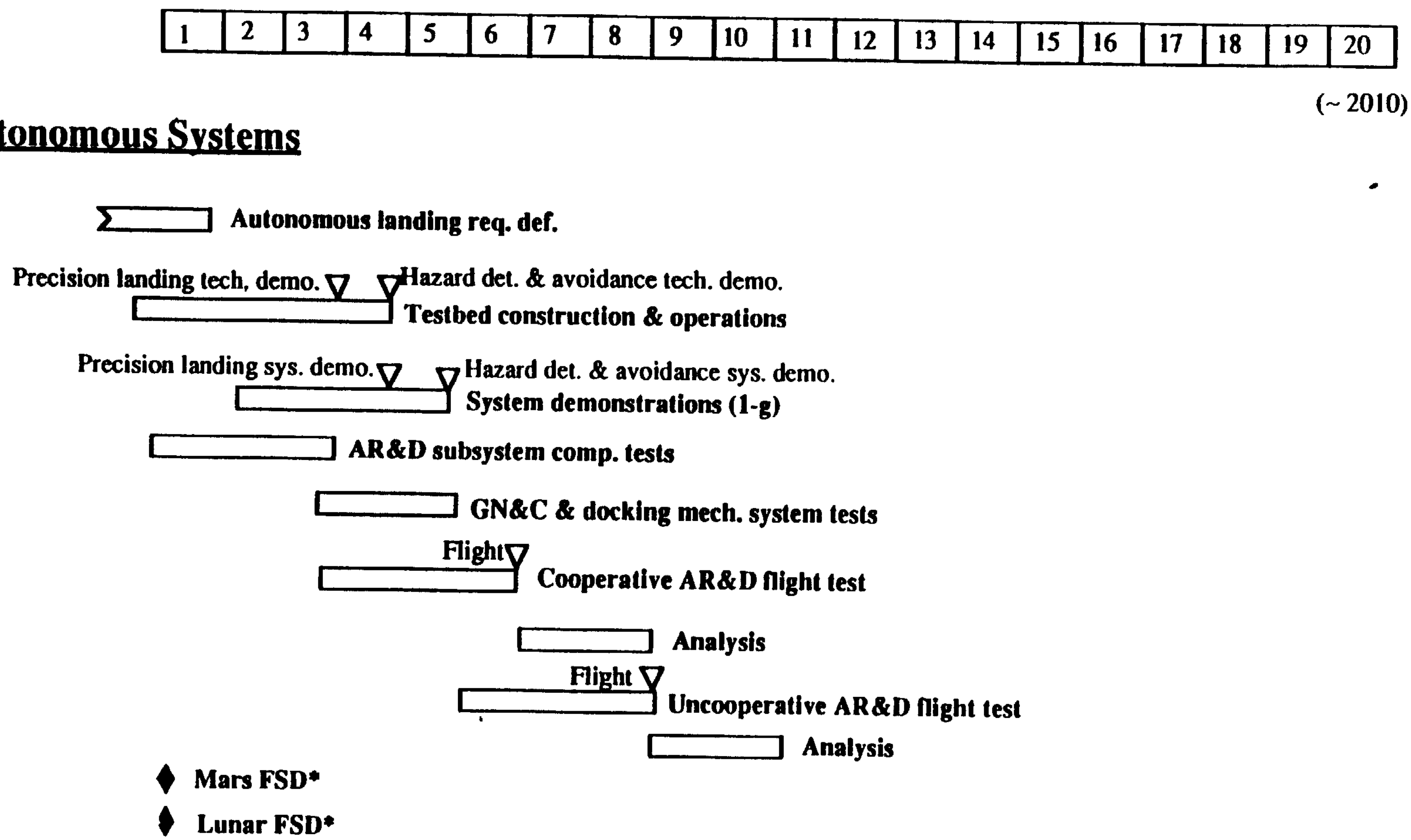
ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

BOEING



* Technology should not present FSD threatening problems; current technologies adequate for minimum mission.

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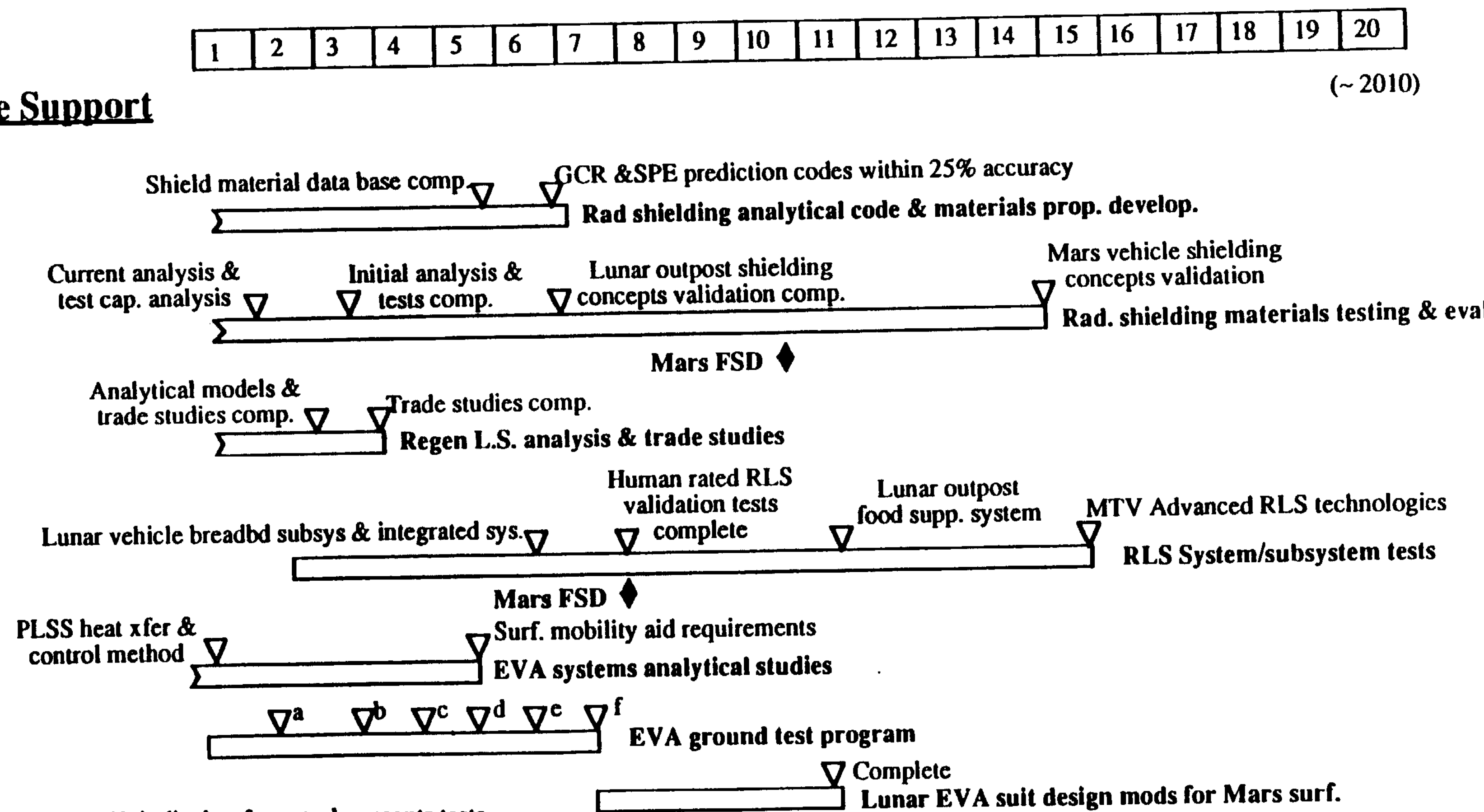
Preliminary SEI Technology Development Schedules (Cont.)

ADVANCED CIVIL SPACE SYSTEMS

BOEING

Life Support



a - Helmet/duit display & control concepts tests
b - Lunar surf suit breadbd test
c - Gloves & displays in simul. SSF environment
d - Regen. PLSS breadbd for lunar surf.
e - Verif tests of adv. dexterious gloves & disp.
f - Complete breadbd lunar EVA suit / simulated surf. cond.

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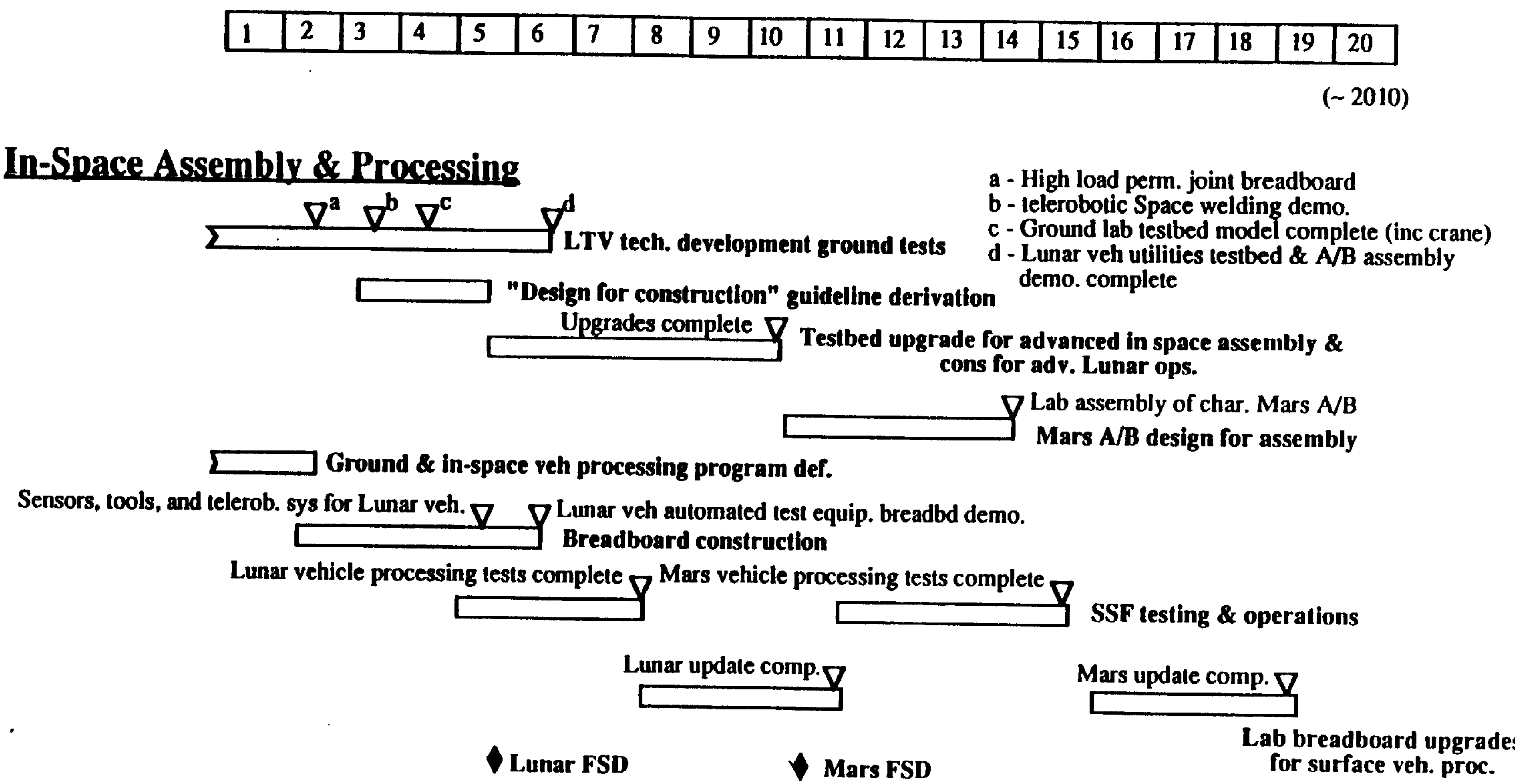
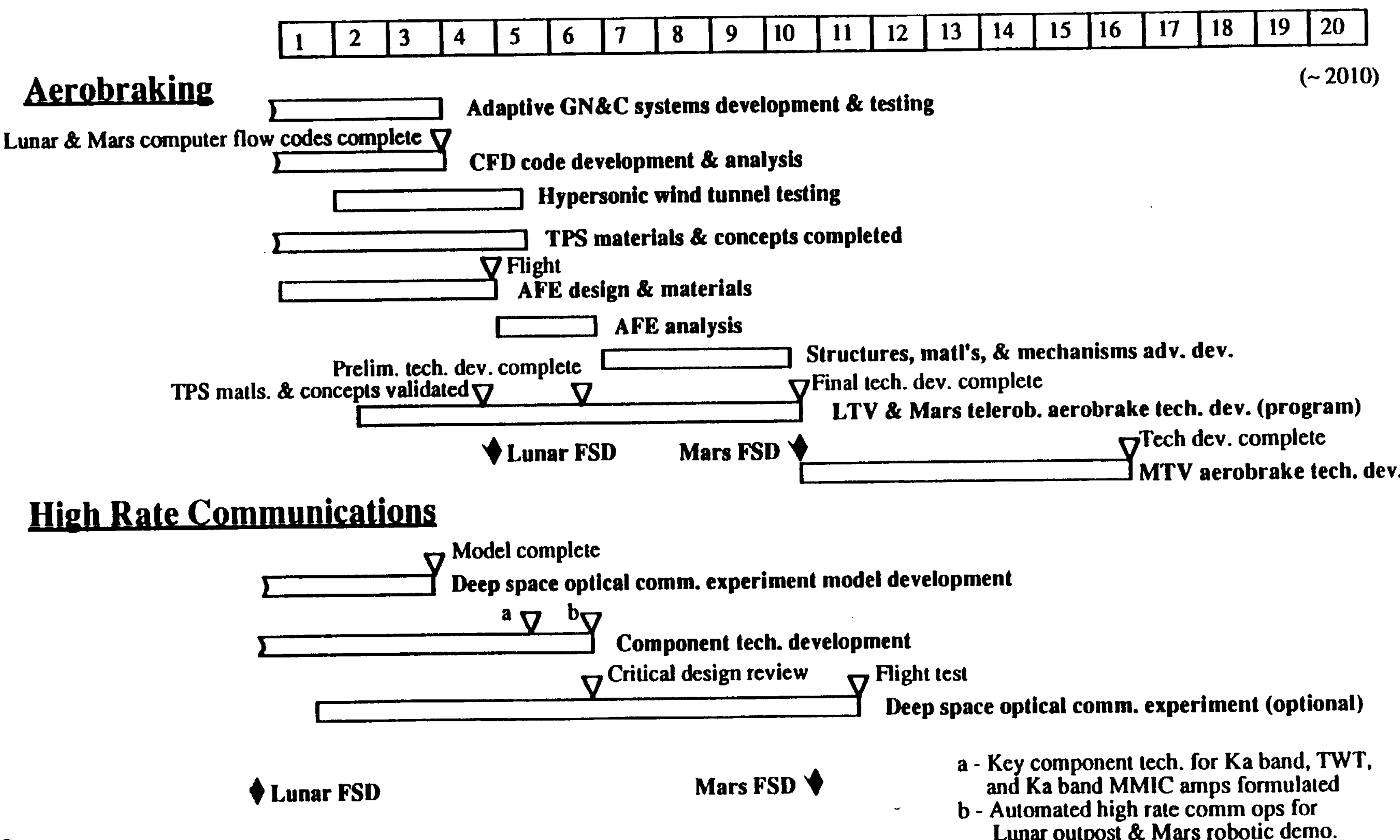
Preliminary SEI Technology Development Schedules (Cont.)



ADVANCED CIVIL SPACE SYSTEMS

Preliminary SEI Technology Development Schedules (Cont.)

BOEING



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Facilities

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Facilities

The facility needs have only been identified in this study; the extent of the impact is yet to be determined. A "bona fide" facility development plan has not been done as some of the requirements are only at a top-level needs evaluation. Therefore, the exact nature of the subsystems and their support facilities are undetermined. When these determinations have been made for the final NASA selected vehicle, the results must be integrated with the vehicle development schedule.

In addition to the information here, additional facility and equipment detail is shown in Ground subsection of the Support Systems section of this text. The volumes for the baseline Cryo/Aerobrake vehicle for assembly, storage, and launch processing are shown in the "Facility Requirements" chart. Processing time shown in the "Assembly Time per Mission" chart. All impacts will be to increase the processing time and working volumes required. Any facility requirements must be viewed in the light of and incorporated into the National Launch Facility Plan.

Facility Requirements

	Assembly Volume	Storage Volume	Launch Processing
1	20694.13	0	0
2	20694.13	0	0
3	42233.11	0	0
4	56989.01	0	0
5	69879.77	10129.05	0
6	54623.87	10129.05	0
7	39222.88	25031.66	4626.85
8	39222.88	25031.66	0
9	49351.93	14902.61	0
10	20694.13	25031.66	18528.75
11	20694.13	34296.04	0
12	20694.13	34296.04	0
13	20694.13	25031.66	9264.38
14	39481.26	25031.66	0
15	39481.26	25031.66	0
16	0	25031.66	16912.13
17	18528.75	25031.66	0
18	18528.75	10129.05	0
19	0	25031.66	18528.75
20	0	34296.04	0
21	0	34296.04	0
22	0	25031.66	9264.38
23	0	25031.66	0
24	0	25031.66	0
25	0	10129.05	14902.61
26	21207.95	10129.05	0
27	21207.95	30387.15	0
28	0	30387.15	21207.95
29	0	30387.15	10129.05
30	0	30387.15	10129.05
31	0	20258.1	10129.05
32	0	20258.1	10129.05
33	0	20258.1	10129.05
34	0	20258.1	10129.05
35	0	10129.05	10129.05
36	0	10129.05	10129.05

Assembly Time per Mission

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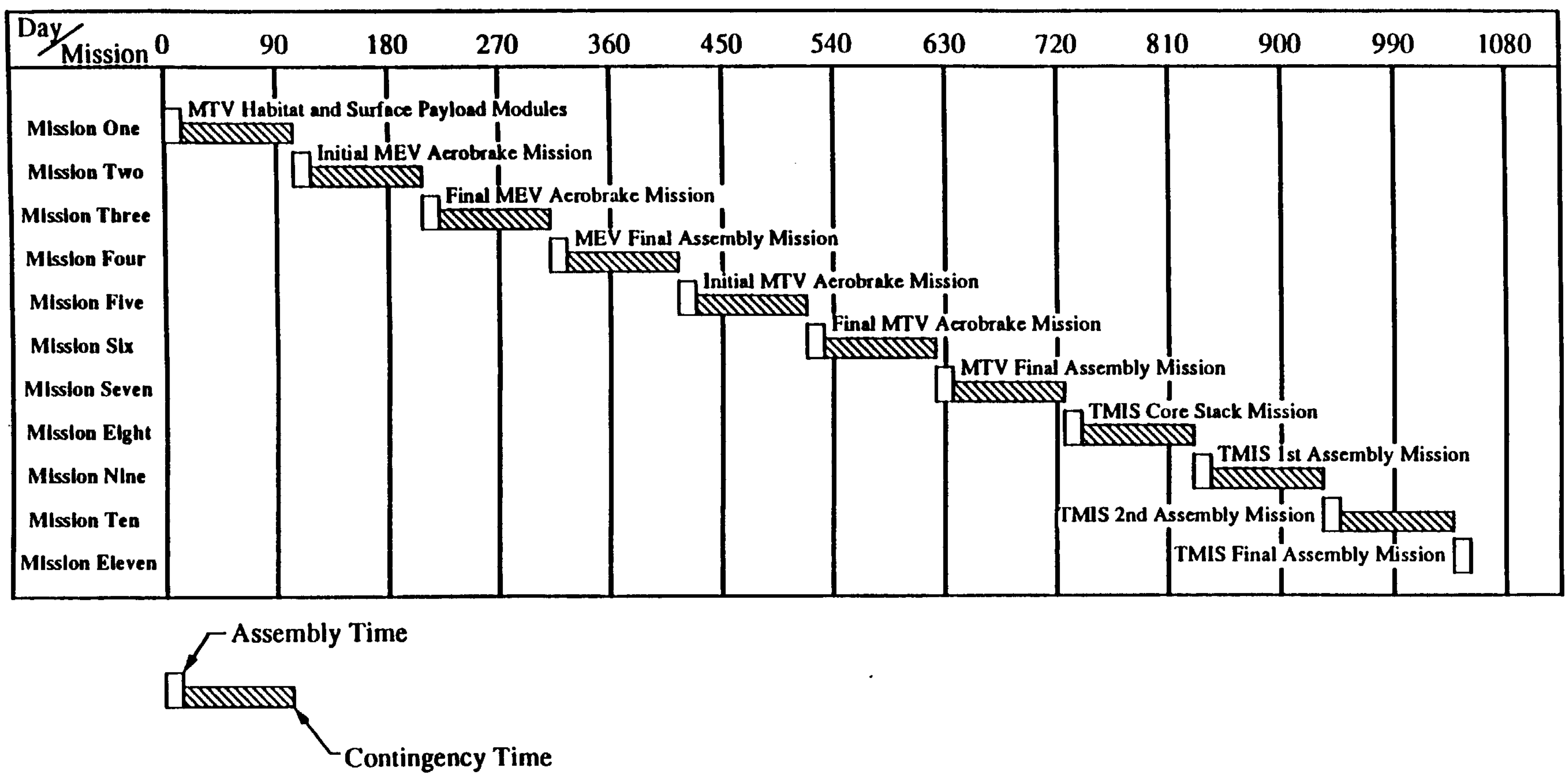

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Costs

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Programmatics

The objectives of the Programmatics task during the current phase of the study were: (1) realistic initial schedules that include initial critical path program elements; (2) initial descriptions of new or unique facilities requirements; (3) development of a stable, clear, responsive work breakdown structure (WBS) and WBS dictionary; (4) initial realistic estimates of vehicle, mission and program costs, cost uncertainties, and funding profile requirements; (5) initial risk analysis, and (6) early and continuing infusion of programmatics data into other study tasks to drive requirements/design/trade decisions.

The issues addressed during the study to date included: (1) capturing all potential long-lead program items such as precursor missions, technology advancement and advanced development, related infrastructure development, support systems and new or modified facility construction, since these are as important as cost and funding in assessing goal achievability; (2) incorporating sufficient operating margin in schedules to obtain high probability of making the relatively brief Mars launch windows; (3) the work breakdown structure must support key study goals such as commonality and (4) cost estimating accuracy and uncertainty are recurring issues in concept definition studies.

Introduction

The study flow, as required by MSFC's statement of work, began with a set of strawman concepts, introduced others as appropriate, conducted "neckdowns", and concluded with a resulting set of concepts and associated recommendations.

As the study progressed, much discussion among the SEI community centered on "architectures". In this study, architectures were more or less synonymous with concepts, since the statement of work required that each concept be fully developed including operations, support, technology, and so forth.

We started with ten concepts as shown in "Overall Study Flow" chart. After the "neckdown" was completed, significant effort was put into programmatics.

As was indicated earlier, we established three levels of activity to evaluate in-space transportation options. The minimum was just enough to meet the President's objectives; in fact "return to the Moon to stay" was interpreted as permanent facilities but not permanent human presence. The minimum program had only three missions to Mars. The median (full science) program aimed at satisfying most of the published science objectives for Lunar and Mars exploration. The maximum program aimed for industrialization of the Moon, for return of practical benefits to Earth, and for the beginnings of colonization of Mars. The range of activity levels, as measured by people and materiel delivered to planetary surfaces, was about a factor of 10. The range of Earth-to-orbit launch rates was less, since we adopted results of preliminary trade studies, selecting more advanced in space transportation technologies as baselines for greater activity levels. The high level schedules developed for these three levels of activity are shown in the "Minimum Program", "Full Science Program" and "Industrialization and Settlement Program" charts and a comparison of them for both Lunar and Mars is shown in the "Lunar Program Comparison" and "Mars Program Comparison" charts. The Cryogenic-All propulsive systems were derived from the Cryo/Aerobrake systems be adjusting the size of the Trans-Mars Injection Stage and eliminating the aerobrake from the materials costed.

Schedule/Network Development Methodology

A PC system called Open Plan by WST Corporation was used, which allows direct control and lower cost over a larger (mainframe) system. The network was purposely kept simple. Summary activities were used in development of the networks. When detailed to a lower level, some activities will require a different calendar than we used. One calendar with a five day work week - no holiday was used. Utilizing multicalendars on a summary network could confuse the development. The Preliminary WBS Structure Level 7 was followed for selection of work to be detailed. An example of Level 7 is: MEV Ascent Vehicle Structure/Mechanisms. We then developed a generic logic string of activities with standard durations for like activities. This logic was then applied against each WBS Level 7 element. To establish interface ties between logic strings and determination of major events, we used the Upper Level Summary Schedule and Summary Level Technology Schedule.

Goals/Purpose

There were two goals for the schedule/network development. These were:

a. Guidelines for Future Development. The schedules are a preliminary road map to follow in the development program.

b. Layout Basis Framework for Network. The networks can be used for future detail network development. This development can be in phases retaining unattended logic for areas which can be be detailed.

Status

Six preliminary networks have been developed. They are:

- Lunar minimum
- Lunar full science
- Lunar industrialization
- Mars missions
- Mars full science
- Mars settlement

These networks will be further developed as information becomes available. The technology development plan schedules are shown in the Schedules subsection of this text ; an example of the standard 6 year program phase C/D schedule is shown in "Reference 6 yr. Full-Scale Development Schedule" chart. The network schedules developed during the study are available in the Final Report Costs Data Book and the WBS.

Facilities

The facility requirements and approaches are discussed in the Facilities section of this text.

Development Implementation

The integrated technology advancement and full-scale development schedules for the Cryogenic/Aerobrake is shown for the subsystems in the Schedule section of this document. The MEV is developed according to the above mentioned standard 6-year FSD schedule. The Man-rating schedules for critical systems, that must be accomplished before first flight, are given in the next six man-rating charts. The long-duration Mars Tansit Habitat, and its critical subsystems, will require operational testing in space to qualify for the Mars mission. How all development and testing is actually done depends on program interrelationships between lunar and Mars missions.

Work Breakdown Structure

The approach to developing a WBS tree and dictionary was to use the Space Station Freedom Work Package One WBS as a point of departure to capture commonality, modularity and evolution potentials. We worked with MSFC to evolve the WBS illustrated in the six WBS charts given in this section. The WBS dictionary details are provided with the WBS tree in a separate deliverable document.

Cost Data

Overall Approach

Space transfer concept cost estimates were developed through parametric and detail estimating techniques using program/scenario plans and hardware and software descriptions combined with NASA and subcontractor data. Our estimating approach simulates the aerospace development and production environment. It also reflects program options not typical of aerospace programs. This flexibility allows assessment of innovative program planning concepts.

Several tools were employed in this analysis. For developing estimates the Boeing Parametric Cost Model (PCM) designed specifically for advanced system estimating was used. It utilizes a company-wide, uniform computerized data base containing historical data compiled since 1969. The second major tool is a Boeing developed Life Cycle Cost Model. The third tool is the Boeing developed Return on Investment (ROI) Analyses.

The approach to cost estimating was to use the PCM to establish DDT&E and manufacturing cost of major hardware components or to use other estimates, (e.g. Nuclear Working Group estimator) if they were considered superior and then feed them to the LCC model. Variations on equipment hardware or mission alternatives can be run through the LCC and then compared for a return on investment. This flow is illustrated in the "Costing Methodology Flow" chart. We were able to investigate alternative concepts quickly, giving system designers more data for evolving scenario/mission responsive concepts. Transportation concepts, trade studies, and "neckdown" efforts were supported by this approach.

Parametric Cost Model

PCM develops cost from the subsystem level and builds upward to obtain total program cost. Costs are estimated from physical hardware descriptions (e.g., weights and complexities) and program parameters (e.g., quantities, learning curves, and integration levels). Known costs are input directly into the estimate when available; the model assesses the necessary system engineering and system test efforts needed for integration into the program. The PCM working unit is man-hours, which allows relationships that tie physical hardware descriptions first to design engineering or basic factory labor, and then through the organizational structure to pick up functional areas such as systems engineering, test, and development shop. Using man-hours instead of dollars for estimating relationships enables more reliable estimates. The PCM features, main inputs, and results are shown in the "Boeing Parametric Cost Model (PCM)" chart. The applicable PCM results, in constant 1990 dollars, are then put into the Life Cycle Cost Model to obtain cost spreads for the various missions/programs. The various hardware components costed for the three different missions/programs are shown in the "LCCM Hardware Assignments" chart. As stated above, adjustments were made for the Cryogenic-All Propulsive from the Cryo/Aerobrake configuration.

The development of space hardware and components needed to accomplish the three different Lunar/Mars missions were identified. These components are grouped into three different categories defined below.

HLLV(Heavy Lift Launch Vehicle) is the booster required to lift personnel, cargo and fuels into LEO and support the LEO node operations.

Propulsion Includes the space propulsion system required to transfer people, cargo and equipment out of LEO and into space. Space means Lunar, Mars and Earth destinations. Propulsion Systems also include an all-propulsive cryogenic Trans Mars Injection System (TMIS) for the Minimum Mission, the Nuclear Electric Propulsion Stage for the Settlement/Industrial Missions.

Modules Include the space systems that are required to transfer people, cargo and equipment from LEO to Lunar and Mars orbit; to de-orbit and sustain life and operations on the Lunar and Mars Surface; and, finally, to return personnel and equipment to LEO.

Cost Buildups

The PCM cost Model can be used directly to obtain complete DDT&E cost, including production of major test articles, by entering into the manufacturing section the equivalent numbers of units for each item, including the first flight article. However, when operated in this way, PCM does not give the first unit cost. To save time, we operated PCM so as to give first unit cost, which we needed for life cycle cost analyses, and used the first unit cost to manually estimate the test hardware content of the DDT&E program. The "wrap factors" shown in the cost buildup sheets were derived from the PCM runs as the factor that is applied to design engineering cost to obtain complete design and development costs, e.g. including non-recurring items such as systems engineering and tooling development.

Life Cycle Cost Model

The LCCM cost data is a composite of HLLV costs, launch base facilities cost estimate based on $/sq. ft. and parametric estimates derived from the Parametric Cost Model. The principal source of information is from the PCM. All hardware cost estimates, with the exception of HLLV, have been developed with this model.

The LCCM consists of three individual models. One model is for the Minimum Program Scale; the second is for the Full Science Program Scale; while the third model is for the Settlement/Industrialization Program Scale. The Minimum Program meets the President's Space Exploration Initiative (SEI) objectives. These capabilities include permanent Lunar facilities but not permanent human presence and three missions to Mars. The Full Science program not only meets the President's SEI objectives but also provides for long term bases for far-ranging surface exploration. The Settlement/Industrialization program accomplishes the objectives of the Minimum and Full Science program scales and additionally returns practical benefits to Earth. These models were developed using the three architecture levels described in the Boeing manifest worksheets. Total cost for each system are tabulated by year and each year's totals feed into a summary sheet that calculates the total program cost for each level. Since the LCCM results are mission related, not just vehicle related, they are not provided here but are available in Final Report Cost Data Book. The LCCM was developed using Microsoft Excel version 2.2 for the Macintosh computer. Any Macintosh equipped with Excel 2.2 can be used to execute the model.

Return On Investment

One of the principal uses of the LCCM is to develop trades and return on investment for technology options. As shown in the "Costing Methodology Flow" chart , two separate life cycle cost models (which include DDT&E and production cost data derived from the parametric cost models) must be developed for each ROI case; a reference, and a case utilizing a technology option. The two life cycle cost streams are separately entered, and the ROI model is executed. The flow also illustrates that not all of the data entered into the life cycle cost model is derived from available costing software. Technical analysis must accompany this data. For example, the number of units which must be produced for the DDT&E program must be determined. This is done at the subsystem level based on knowledge of past programs, and proposed system/subsystem tests. Since the ROI analysis is mission related, not just vehicle related, the data is not presented here but is available in the Final Report Costs Data Book .

Results

A summary of the cost data produced by the PCM for the CAB vehicle are given in the "Mars CAB Preliminary PCM Summary" and "Mars CAB Preliminary PCM Summary - continued" charts . The PCM program was used to produce DDT&E and production cost estimates for each of our reference Mars and lunar vehicles to the subsystem level. The DDT&E costs generated by the PCM do not include all of the necessary hardware for the first mission vehicle. Hence all necessary additional units (prototypes,test units, lab units, etc.) were added into the vehicle cost buildups as shown in the "CAB Cost Buildup" charts. The total DDT&E includes additional costs (e.g.. additional units in the DDT&E program), contractor fees and the engineering wrap factor. The total DDT&E from the cost buildup and the unit cost from the PCM are the primary vehicle cost inputs to the LCC model

Risk Analyses

Risk analyses were conducted to develop an initial risk assessment for the various architectures. This presentation of risk analysis results considers development risk, man-rating requirements, and several aspects of mission and operations risk.

Development Risk

All of the architectures and technologies investigated in this study incur some degree of development risk; none are comprised entirely of fully developed technology. Development risks are correlated directly with technological uncertainties. We identified the following principal risks:

Cryogenics - High-performance insulation systems involve a great many layers of multi-layer insulation (MLI), and one or more vapor-cooled shields. Analyses and experiments have indicated the efficacy of these, but demonstration that such insulation systems can be fabricated at light weight, capable of surviving launch g and acoustics loads, remains to be accomplished. In addition, there are issues associated with propellant transfer and zero-g gauging. These, however, can be avoided for early lunar systems by proper choice of configuration and operations, e.g. the tandem-direct system recommended elsewhere in this report. This presents the opportunity to evolve these technologies with operations of initial flight systems.

Engines - There is little risk of being able to provide some sort of cryogenic engine for lunar and Mars missions. The RL- 10 could be modified to serve with little risk; deep throttling of this engine has already been demonstrated on the test stand. The risk of developing more advanced engines is also minimal. An advanced development program in this area serves mainly to reduce development cost by pioneering the critical features prior to full-scale development.

Aerocapture and aerobraking - There are six potential functions, given here in approximate ascending order of development risk: aero descent and landing of crew capsules returning from the Moon, aerocapture to low Earth orbit of returning reusable lunar vehicles, landing of Mars excursion vehicles from Mars orbit, aero descent and landing of crew capsules returning from Mars, aerocapture to low Earth orbit of returning Mars vehicles, and aerocapture to Mars orbit of Mars excursion and Mars transfer vehicles. The "Development Risk for Aerobraking by Function chart provides a qualitative development risk comparison for these six functions.

Aerocapture of vehicles requires large aerobrakes. For these to be efficient, low mass per unit area is required, demanding efficient structures made from very high performance materials as well as efficient, low mass thermal protection materials. By comparison, the crew capsules benefit much less from high performance structures and TPS.

Launch packaging and on-orbit assembly of large aerobrakes presents a significant development risk that has not yet been solved even in a conceptual design sense. Existing concepts package poorly or are difficult to assemble or both. While the design challenge can probably be met, aerobrake assembly is a difficult design and development challenge, representing an important area of risk.

Nuclear thermal rockets - The basic technology of nuclear thermal rockets was developed and demonstrated during the 1960s and early 1970s. The development risk to reproduce this technology is minimal, except in testing as described below. Current studies are recommending advances in engine performance, both in specific impulse (higher reactor temperature) and in thrust-to-weight ratio (higher reactor power density). The risks in achieving these are modest inasmuch as performance targets can be adjusted to technology performance.

Reactor and engine tests during the 1960s jetted hot, slightly radioactive hydrogen directly into the atmosphere. Stricter environmental controls since that time prohibit discharge of nuclear engine effluent into the atmosphere. Design and development of full containment test facilities presents a greater development risk than obtaining the needed performance from nuclear reactors and engines. Full- containment facilities will be required to contain all the hydrogen effluent, presumably oxidize it to water, and remove the radioactivity.

Electric Propulsion Power Management and Thrusters - Power management and thrusters are common to any electric propulsion power source (nuclear, solar, or beamed power). Unique power management development needs for electric propulsion are (1) minimum mass and long life, (2) high power compared to space experience, i.e. megawatts instead of kilowatts, (3) fast arc suppression for protection of thrusters. Minimizing mass of power distribution leads to high distribution voltage and potential problems with plasma losses, arcing, and EMI. Thus while power management is a mature technology, the unique requirements of electric propulsion introduce a number of development risks beyond those usually experienced in space power systems.

Electric thruster technology has been under development since the beginning of the space program. Small thrusters are now operational, such as the resistance-heat-augmented hydrazine thrusters on certain communications spacecraft. Small arc and ion thrusters are nearing operational use for satellite stationkeeping.

Space transfer demands on electric propulsion performance place a premium on high power in the jet per unit mass of electric propulsion system. This in turn places a premium on thruster efficiency; power in the jet, not electrical power, propels spaceships. Space transfer electric propulsion also requires specific impulse in the range 5000 to 10,000 seconds. Only ion thrusters and magnetoplasmadynamic (MPD) arc thrusters can deliver this performance. Ion thrusters have acceptable efficiency but relatively low power per unit of ion beam emitting area. MPD thruster technology can deliver the needed Isp with high power per thruster, but has not yet reached efficiencies of interest. Circular ion thrusters have been built up to 50 cm diameter, with spherical segment ion beam grids. These can absorb on the order of 50 kWe each. A 10 MWe system would need 200 operating thrusters. The development alternatives all have significant risk: (1) Advance the state of the art of MPD thrusters to achieve high efficiency; (2) Develop propulsion systems with large numbers of thrusters and control systems; or (3) Advance the state of the art of ion thrusters to much larger size per thruster.

Nuclear power for electric propulsion - Space power reactor technology now under development (SP-100) may be adequate; needed advances are modest. Advanced power conversion systems are required to obtain power-to-mass ratios of interest. The SP-100 baseline is thermoelectric, which has no hope of meeting propulsion system performance needs. The most likely candidates are the closed Brayton (gas) cycle and the potassium Rankine (liquid/vapor) cycle. (Potassium provides the best match of liquid/vapor fluid properties to desired cycle temperatures.) Stirling cycle, thermionics, and a high-temperature thermally-driven fuel cell are possibilities. The basic technology for Brayton and Rankine cycles are mature; both are in widespread industrial use. Prototype space power Brayton and Rankine turbines have run successfully for thousands of hours in laboratories. The development risk here is that these are very complex systems; there is no experience base for coupling a space power reactor to a dynamic power conversion cycle;

there is no space power experience base at the power levels needed; and these systems, at power levels of interest for SEI space transfer application, are large enough to require in-space assembly and checkout. Space welding will be required for fluid systems assembly.

Solar power for space transfer propulsion - Solar power systems for space propulsion must attain much higher power-to-mass ratios than heretofore achieved. This implies a combination of advanced solar cells, probably multi-band-gap, and lightweight structural support systems. Required array areas are very large. Low-cost arrays, e.g. $100/watt, are necessary for affordable system costs, and automated construction of the large area structures, arrays, and power distribution systems appears also necessary. Where the nuclear electric systems are high development risk because of complexity and the lack of experience base at relevant power levels and with the space power conversion technologies, most of the solar power risk appears as technology advancement risk. If the technology advancements can be demonstrated, development risk appears moderate.

Avionics and software - Avionics and software requirements for space transfer systems are generally within the state of the art. New capability needs are mainly in the area of vehicle and subsystem health monitoring. This is in part an integration problem, but new techniques such as expert and neural systems are likely to play an important role.

An important factor in avionics and software development is that several vehicle elements having similar requirements will be developed, some concurrently. A major reduction in cost and integration risk for avionics can be achieved by advanced development of a "standard" avionics and software suite, from which all vehicle elements would depart.

Further significant cost savings are expected from advancements in software development methods and environments.

Environmental Control and Life Support (ECLS) - The main development risk in ECLS is for the Mars transfer habitat system. Other SEI space transfer systems have short enough operating durations that shuttle and Space Station Freedom ECLS system derivatives will be adequate. The Mars transfer requirement is for a highly closed physio-chemical system capable of 3 years' safe and dependable operation without resupply from Earth. The development risk arises from the necessity to demonstrate long life operation with high confidence; this may be expensive in cost and development schedule.

Man-Rating Approach

Man-rating includes three elements: (1) Design of systems to manned flight failure tolerance standards, (2) Qualification of subsystems according to normal man-rating requirements, and (3) Flight demonstration of critical performance capabilities and functions prior to placing crews at risk. Several briefing charts follow: the first summarizes a recommended approach and lists the subsystems and elements for which man-rating is needed; subsequent charts present recommended man-rating plans.

Mission and Operations Risk

These risk categories include Earth launch, space assembly and orbital launch, launch windows, mission risk, and mitigation of ionizing radiation and zero-g risks.

Earth launch - The Earth launch risk to in-space transportation is the risk of losing a payload because of a launch failure. Assembly sequences are arranged to minimize the impact of a loss, and schedules include allowances for one make-up launch each mission opportunity.

Assembly and Orbital Launch Operations - Four sub-areas are covered: assembly, test and on-orbit checkout, debris, and inadvertent re-entry.

Assembly operations risk is reduced by verifying interfaces on the ground prior to launch of elements. Assembly operations equipment such as robot arms and manipulators will undergo space testing at the node to qualify critical capabilities and performance prior to initiating assembly operations on an actual vehicle.

Assembly risk varies widely with space transfer technology. Nuclear thermal rocket vehicles appear to pose minimum assembly risk; cryo/aerobraking are intermediate, and nuclear and solar electric systems pose the highest risk.

Test and on-orbit checkout must deal with consequences of test failures and equipment failures. This risk is difficult to quantify with the present state of knowledge. Indications are: (1) large space transfer systems will experience several failures or anomalies per day. Dealing with failures and anomalies must be a routine, not exceptional, part of the operations or the operations will not be able to launch space transfer systems from orbit; (2)

vehicles must have highly capable self-test systems and must be designed for repair, remove and replace by robotics where possible and for ease of repair by people where robotics cannot do the job; (3) test and on-orbit checkout will run concurrently with propellant loading and launch countdowns. These cannot take place on Space Station Freedom. Since the most difficult part of the assembly, test and checkout job must take place off Space Station Freedom the rest of the job probably should also.

Orbital debris presents risk to on-orbit operations. Probabilities of collision are large for SEI-class space transfer systems in low Earth orbit for typical durations of a year or more. Shielding is mandatory. The shielding should be designed to be removed before orbital launch and used again on the next assembly project.

Creation of debris must also be dealt with. This means that (1) debris shielding should be designed to minimize creation of additional debris, especially particles of dangerous size, and (2) operations need to be rigorously controlled to prevent an inadvertent loss of tools and equipment that will become a debris hazard.

Inadvertent re-entry is a low but possible risk. Some of the systems, especially electric propulsion systems, can have very low ballistic coefficient and therefore rapid orbital decay rate. Any of the SEI space transfer systems will have moderately low ballistic coefficient when not loaded with propellant. While design details are not far enough along to make a quantitative assessment, parts of these vehicles would probably survive reentry to become ground impact hazards in case of inadvertent reentry. For nuclear systems, it will be necessary to provide special support systems and infrastructure to drive the probability of inadvertent reentry to extremely low levels.

Launch Windows - Launch windows for single-burn high-thrust departures from low Earth orbit are no more than a few days because regression of the parking orbit line of nodes causes relatively rapid misalignment of the orbit plane and departure vector. For lunar missions, windows recur at about 9-day intervals.

For Mars, the recurrence is less frequent, and the interplanetary window only lasts 30 to 60 days. It is important to enable Mars launch from orbit during the entire interplanetary window. Three-impulse Mars departures make this possible; a plane change at apogee of the intermediate parking orbit provides alignment with the departure vector. Further analysis of the three-burn scheme is needed to assess penalties and identify circumstances where it does not work.

Launch window problems are generally minimal for low-thrust (electric propulsion) systems.

Mission Risk - Comparative mission risk was analyzed by building risk trees and performing semi-quantitative analysis. The next chart presents a comparison of several mission modes; after that are the risk trees for these modes.

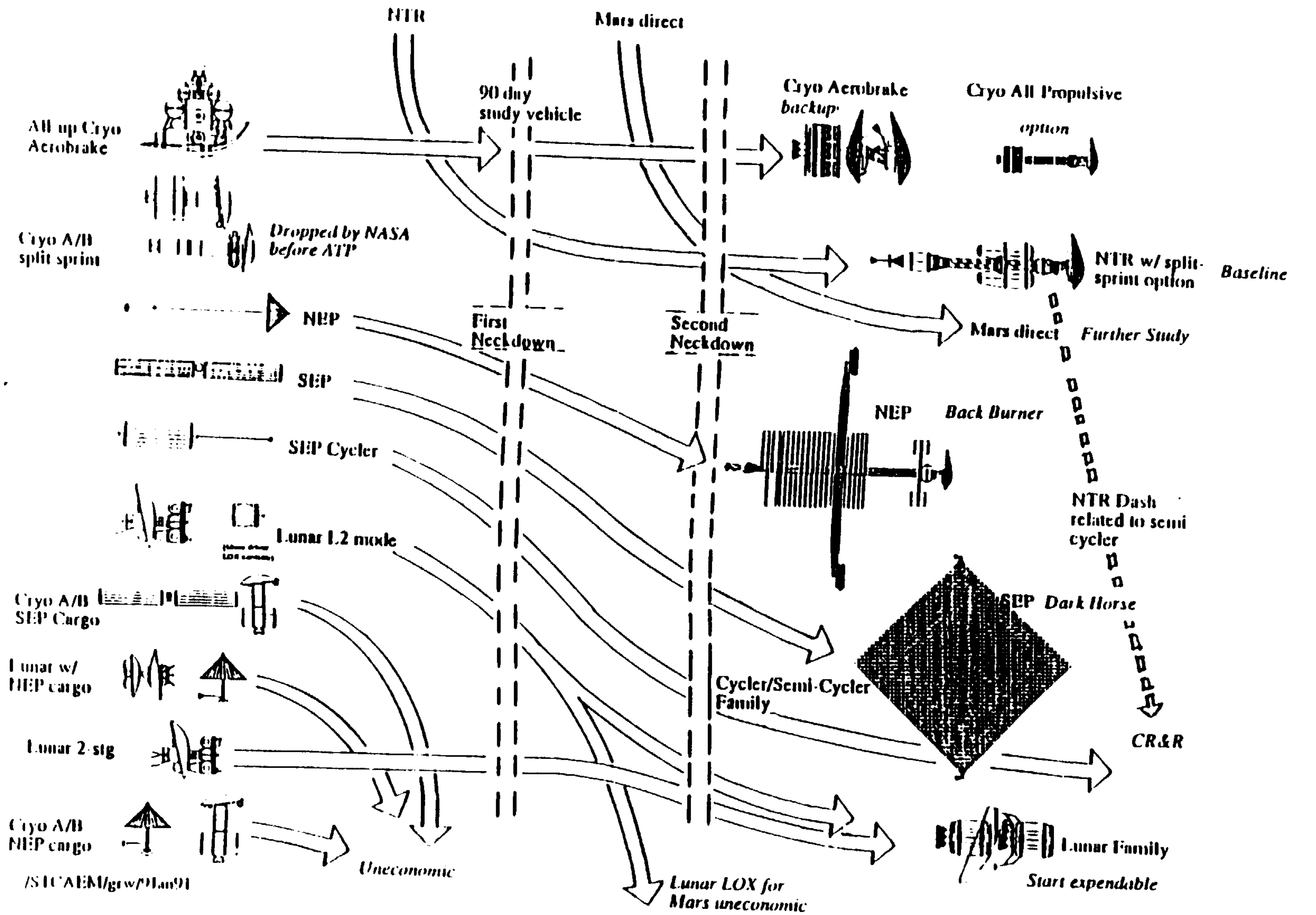
Ionizing Radiations and Zero G - The threat from ionizing radiations is presented elsewhere in this document. Presented here are the mitigating strategies for ionizing radiations and zero g.

Nuclear systems operations present little risk to flight crews. Studies by University of Texas at Austin showed that radiation dose to a space station crew from departing nuclear vehicles is very small provided that sensible launch and flight strategies are used. On-board crews are protected by suitable shielding and by arrangement of the vehicle, i.e. hardware and propellant between reactors and the crew and adequate separation distances. After nuclear engines are shut off, radiation levels drop rapidly so that maneuvers such as departure or return of a Mars excursion vehicle are not a problem. On-orbit operations around a returned nuclear vehicle are deferred until a month or two after shutdown, by which time radioactivity of the engine is greatly reduced.

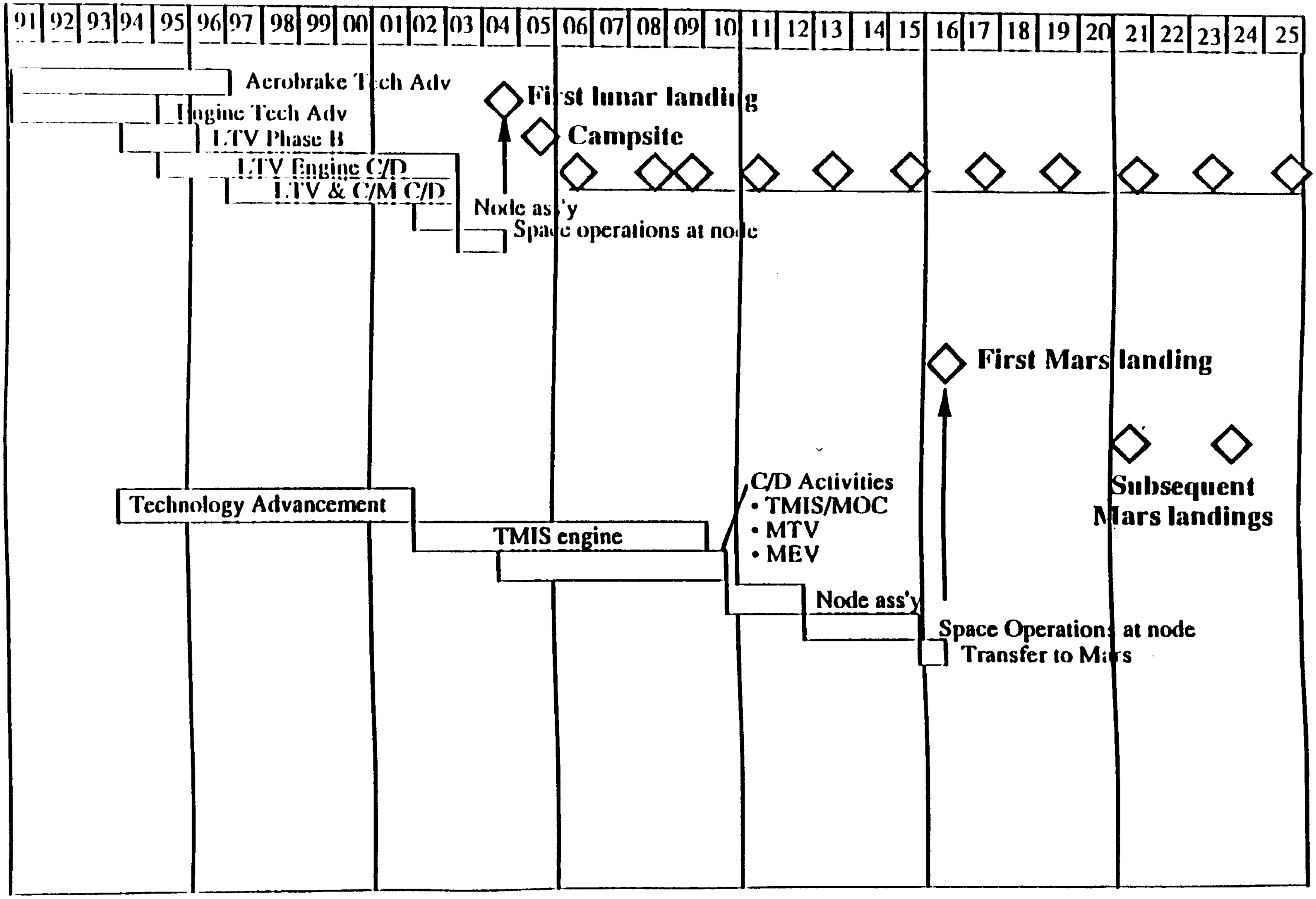
Reactor disposal has not been completely studied. Options include solar system escape and parking in stable heliocentric orbits between Earth and Venus.

Crew radiation dose abatement employs "storm shelters" for solar flares, and either added shielding of the entire vehicle or fast transfers (or both) to reduce galactic cosmic ray exposure. Assessments are in progress; tradeoffs of shielding versus fast trips have yet to be completed. Expected impact for lunar missions is negligible and for Mars missions, modest.

Overall Study Flow



Minimum Program



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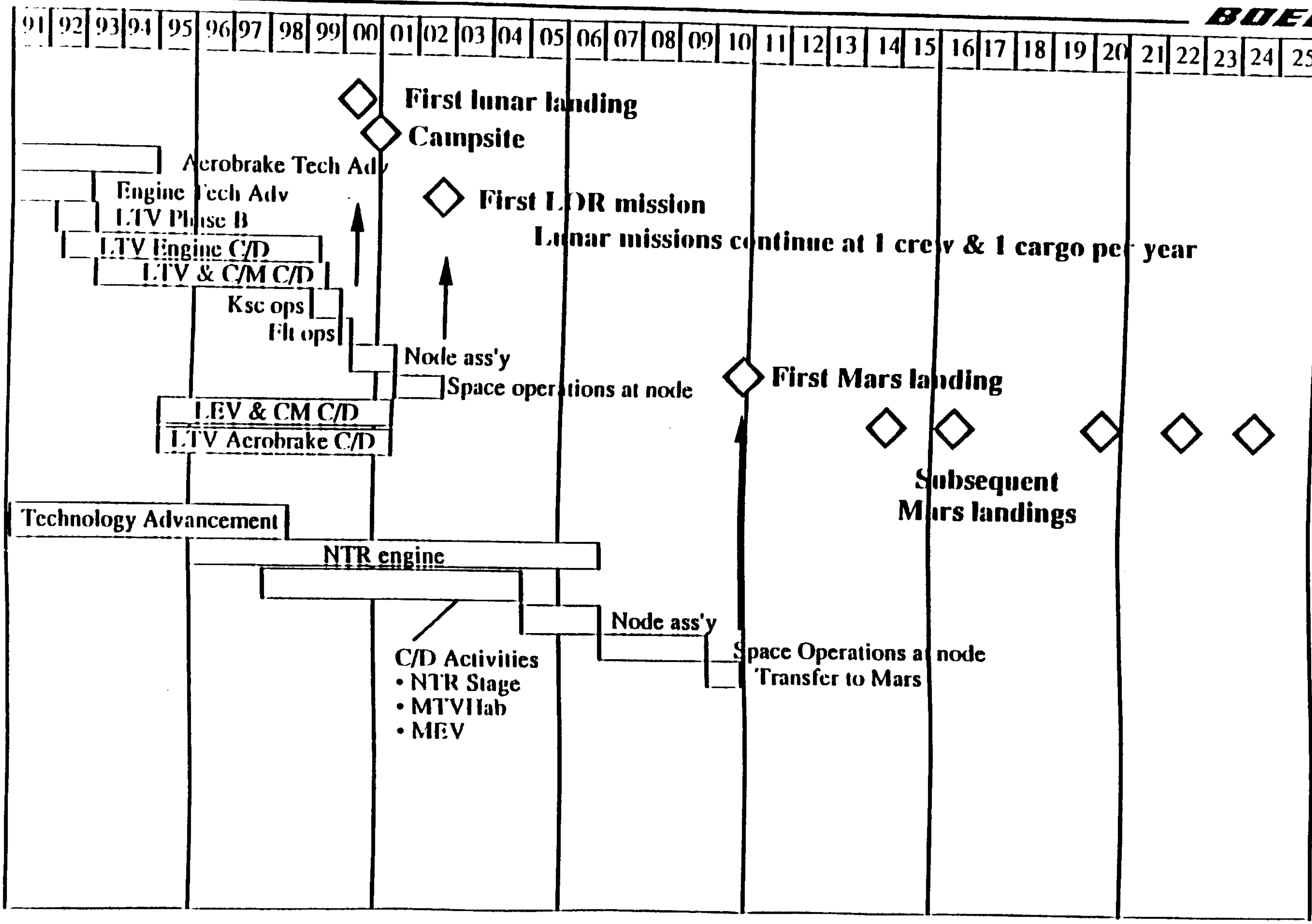
Full Science Program



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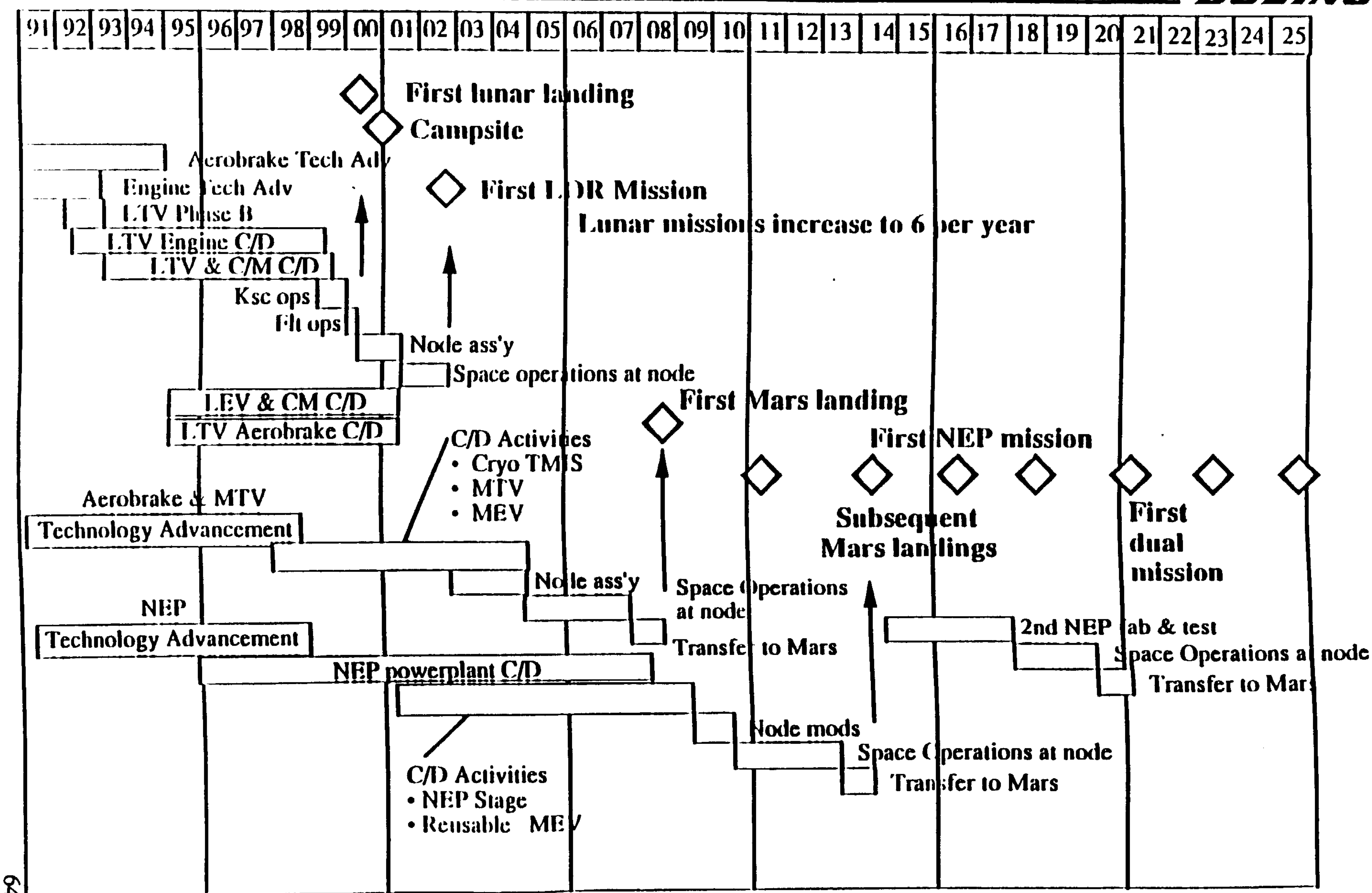
Industrialization and Settlement Program



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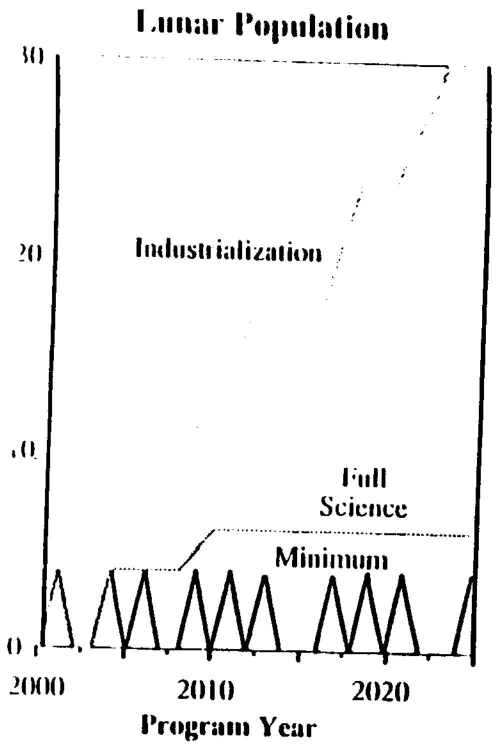
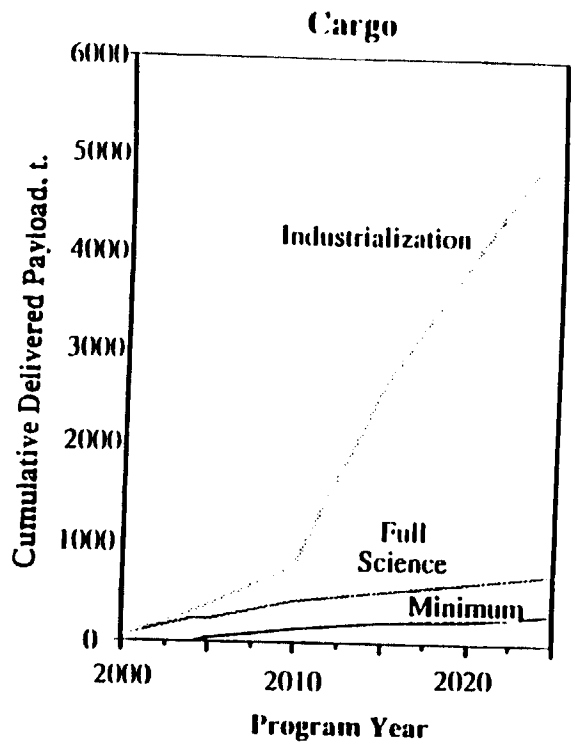
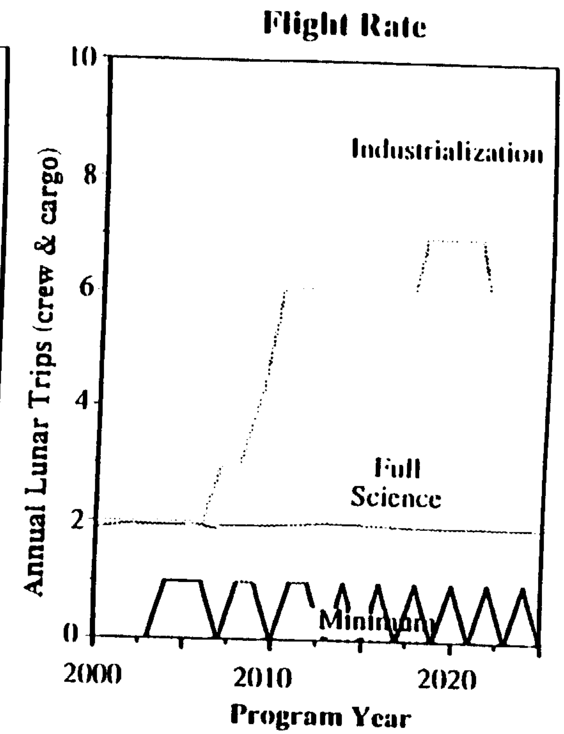
Lunar Program Comparison







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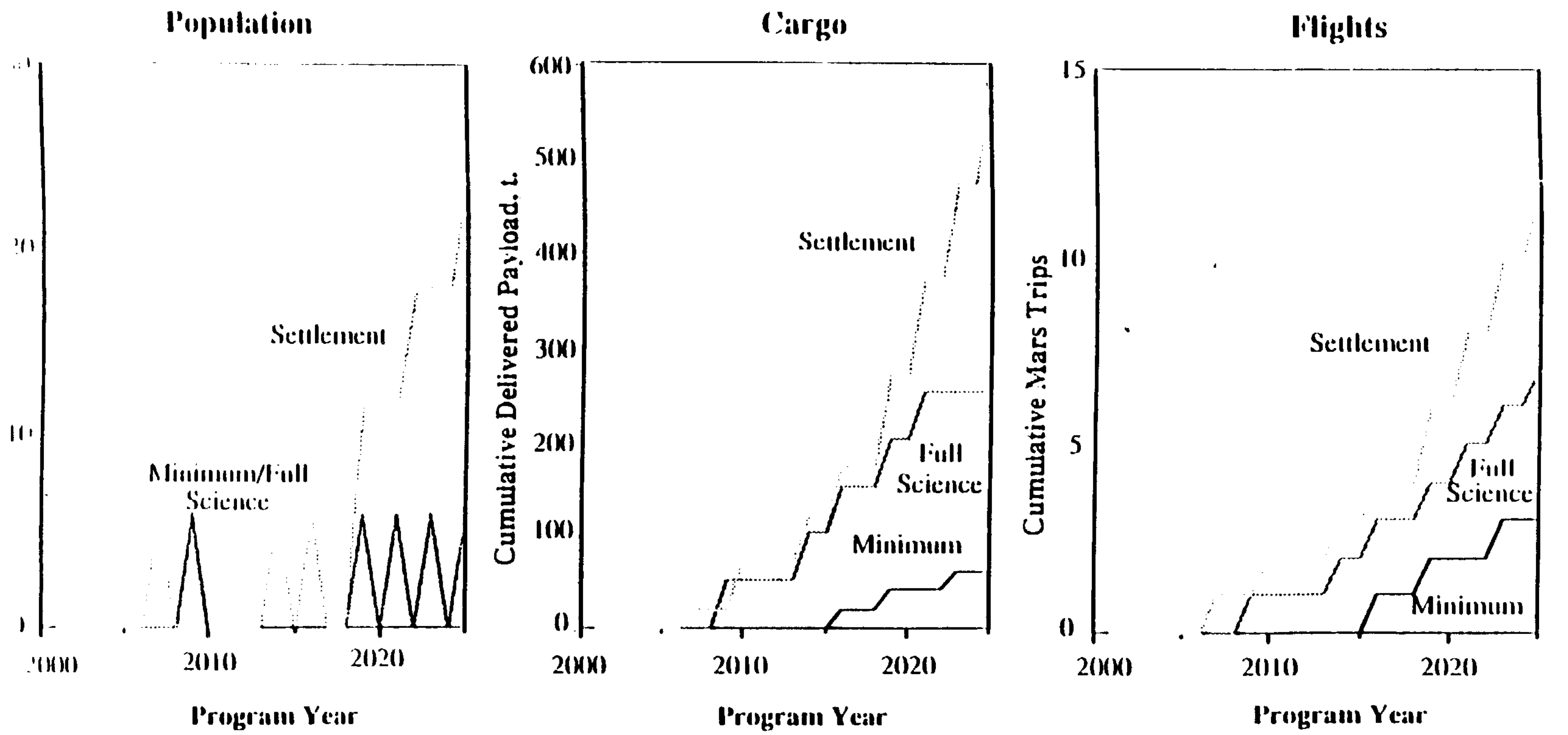
Mars Program Comparisons



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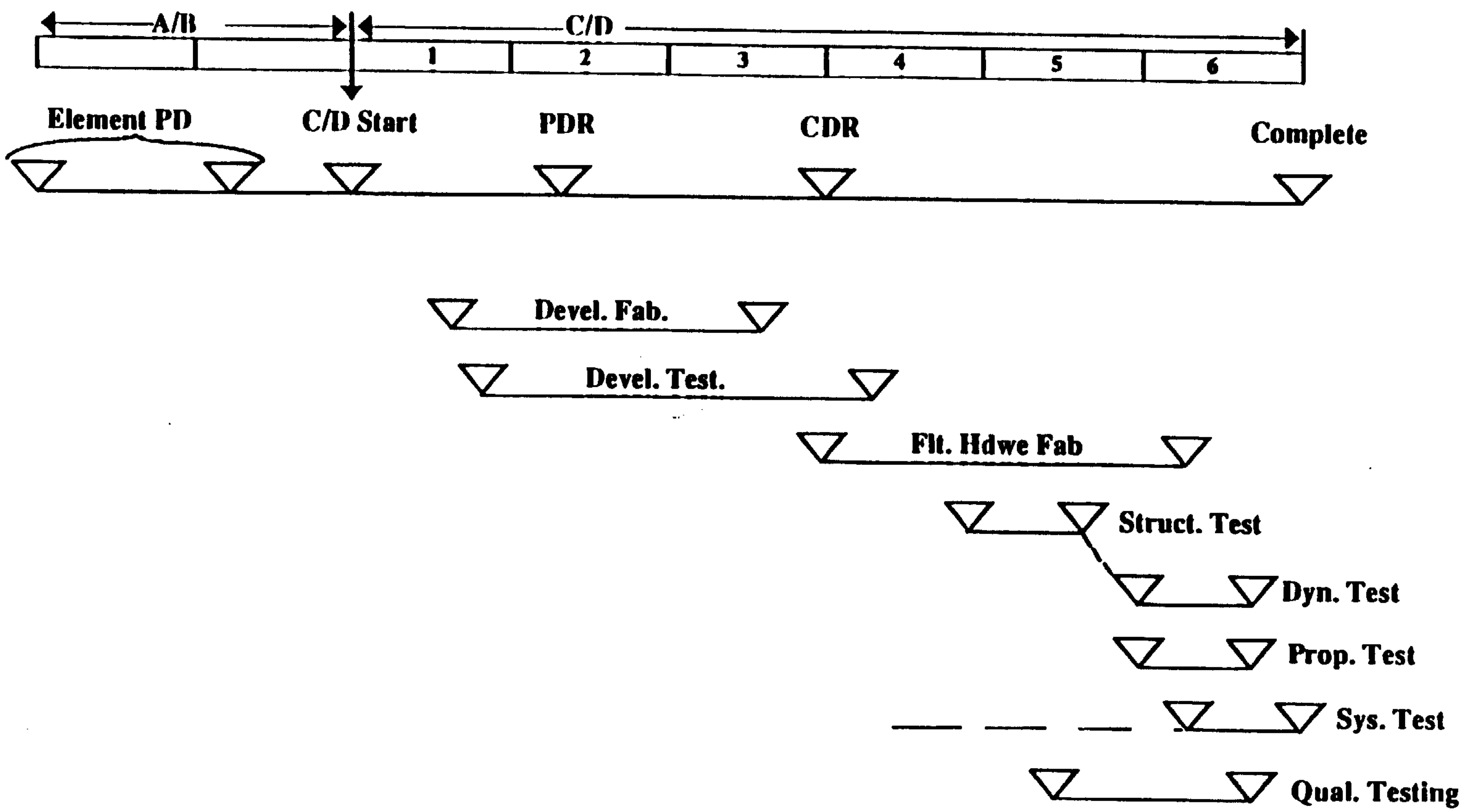
ADVANCED CIVIL SPACE SYSTEMS

Reference 6 yr Full-Scale Development Schedule

BOEING



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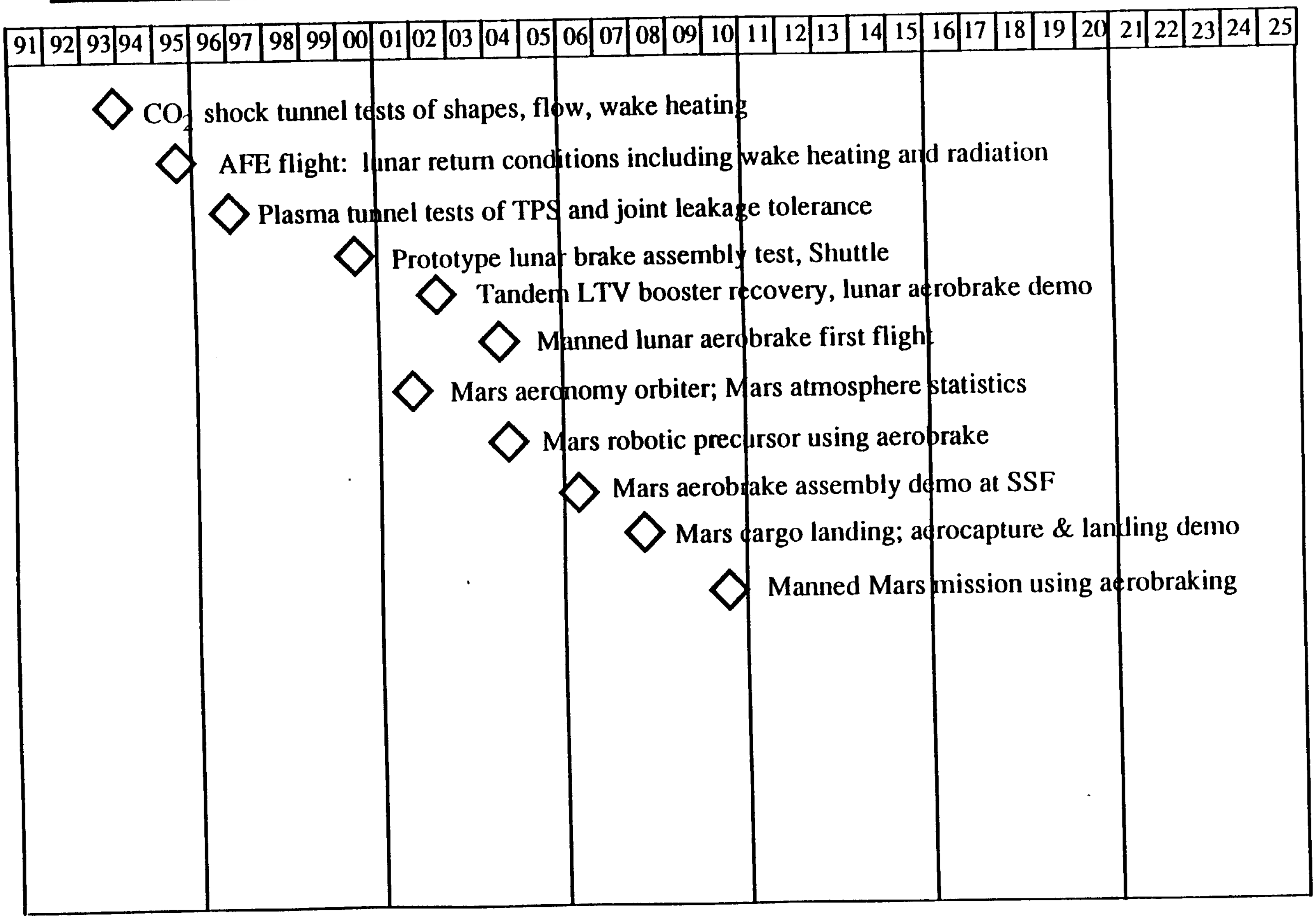
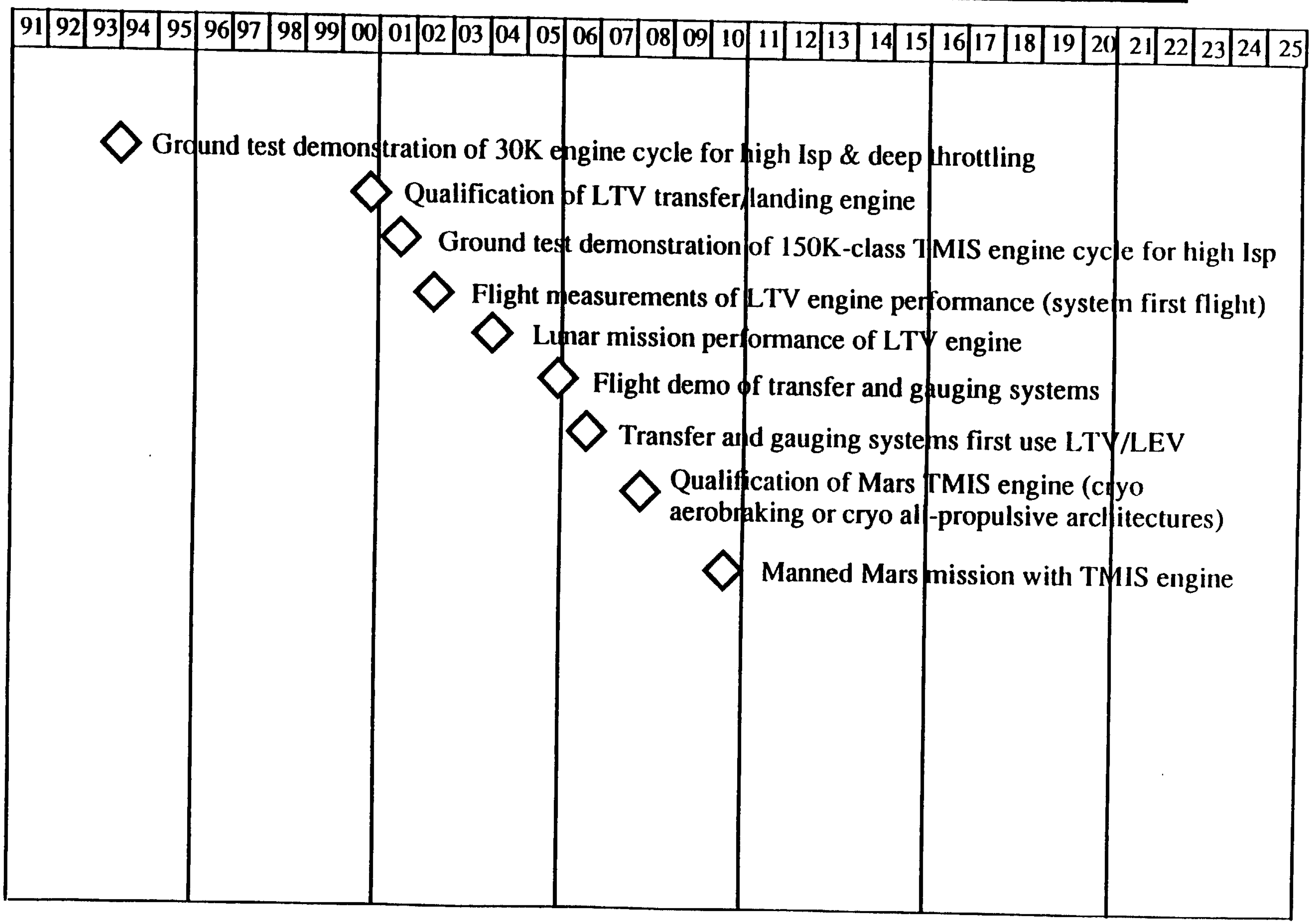
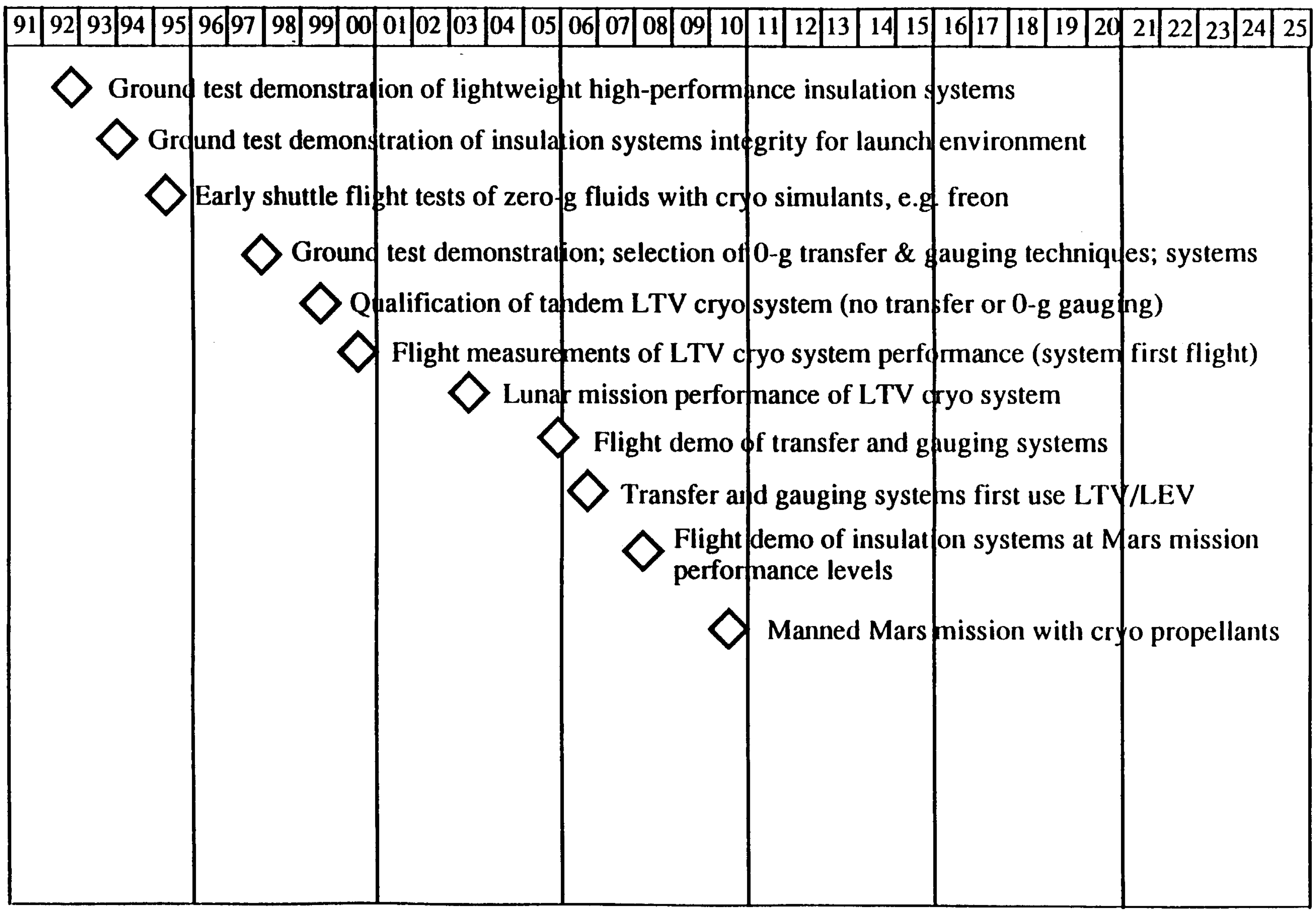
Aerobraking Major Test/Demo Man-Rating Approach



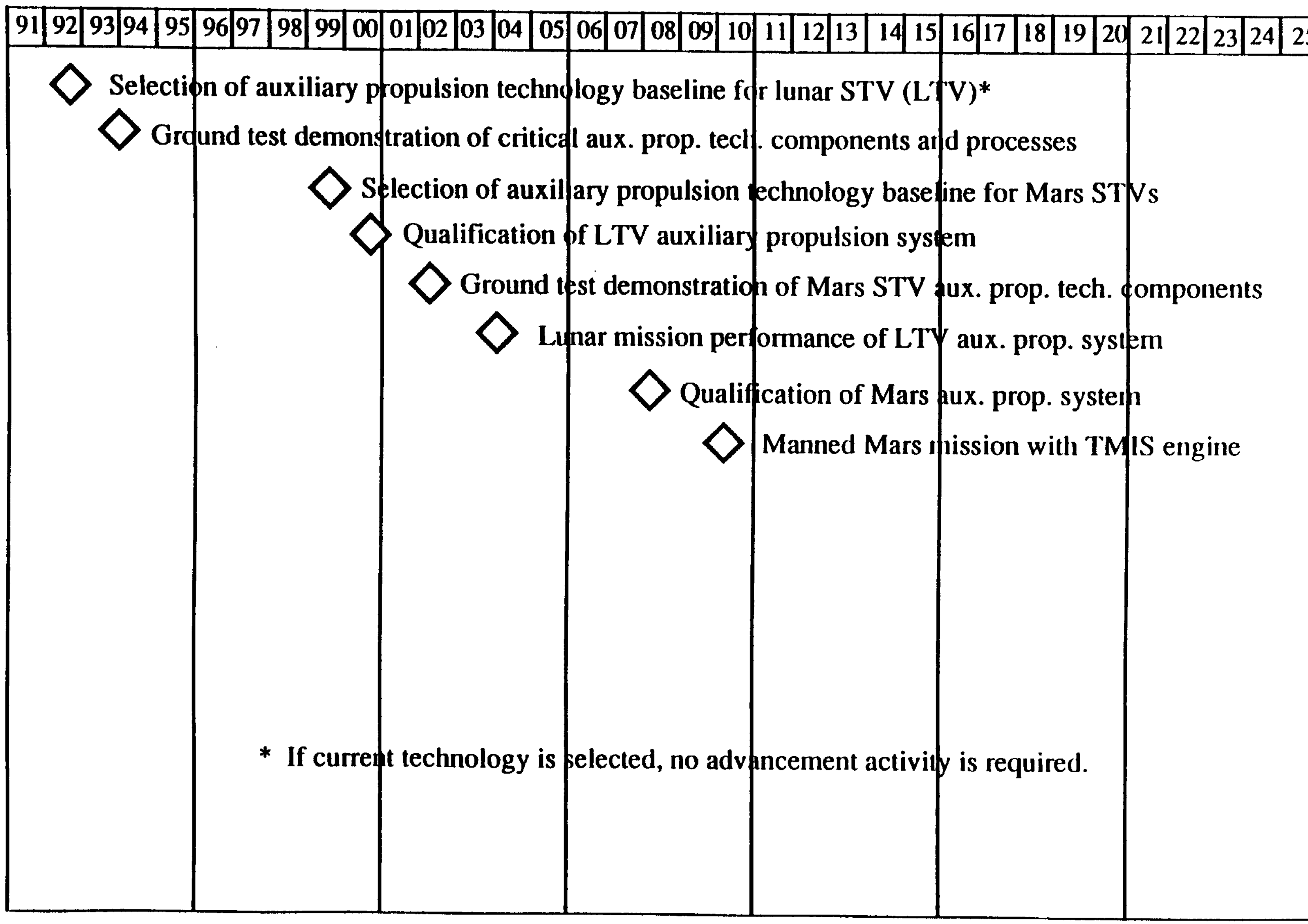
Cryogenic Rocket Engine Man-Rating Approach



Cryogenic Propellant System Man-Rating Approach

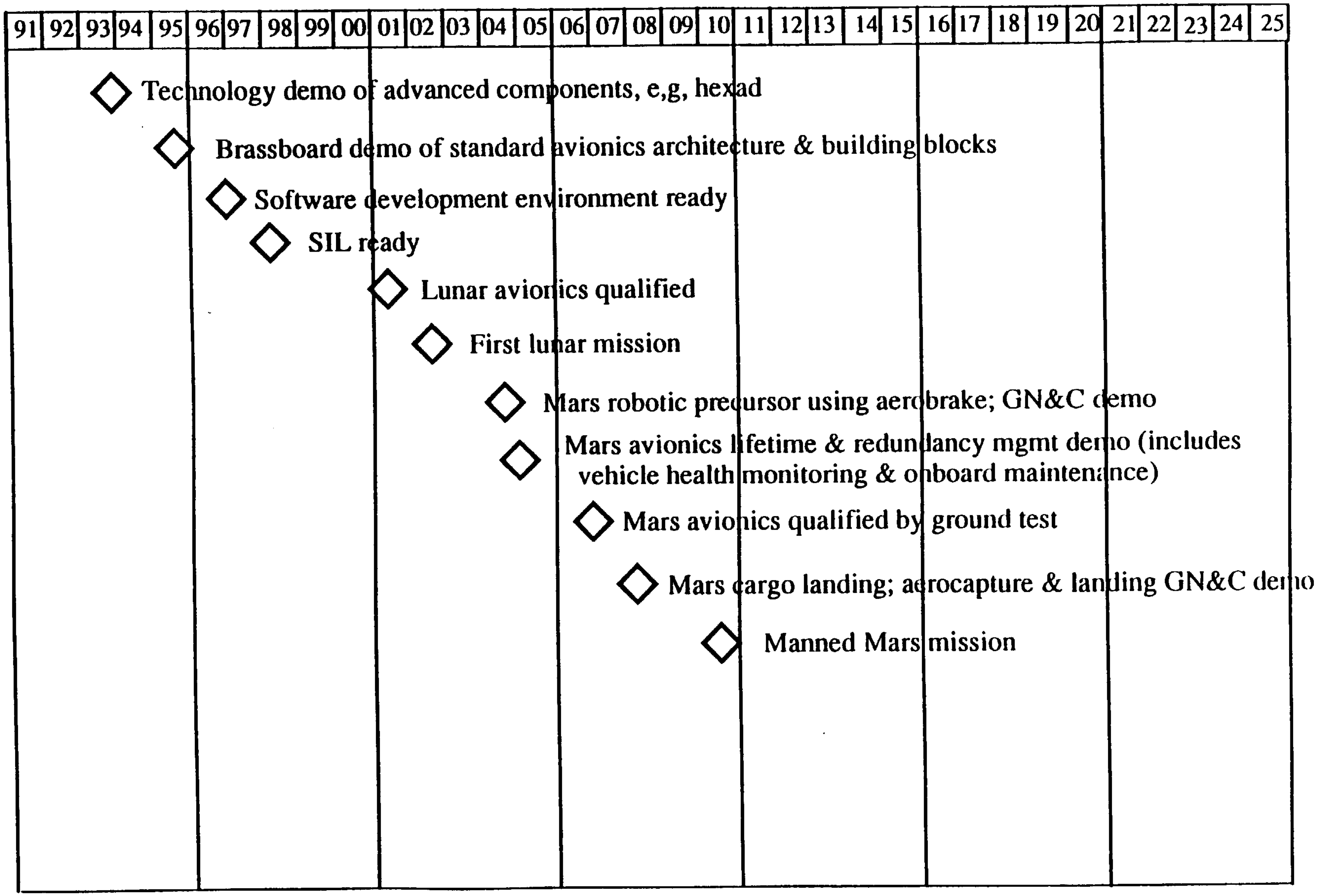


Advanced Auxiliary Propulsion Man-Rating Approach

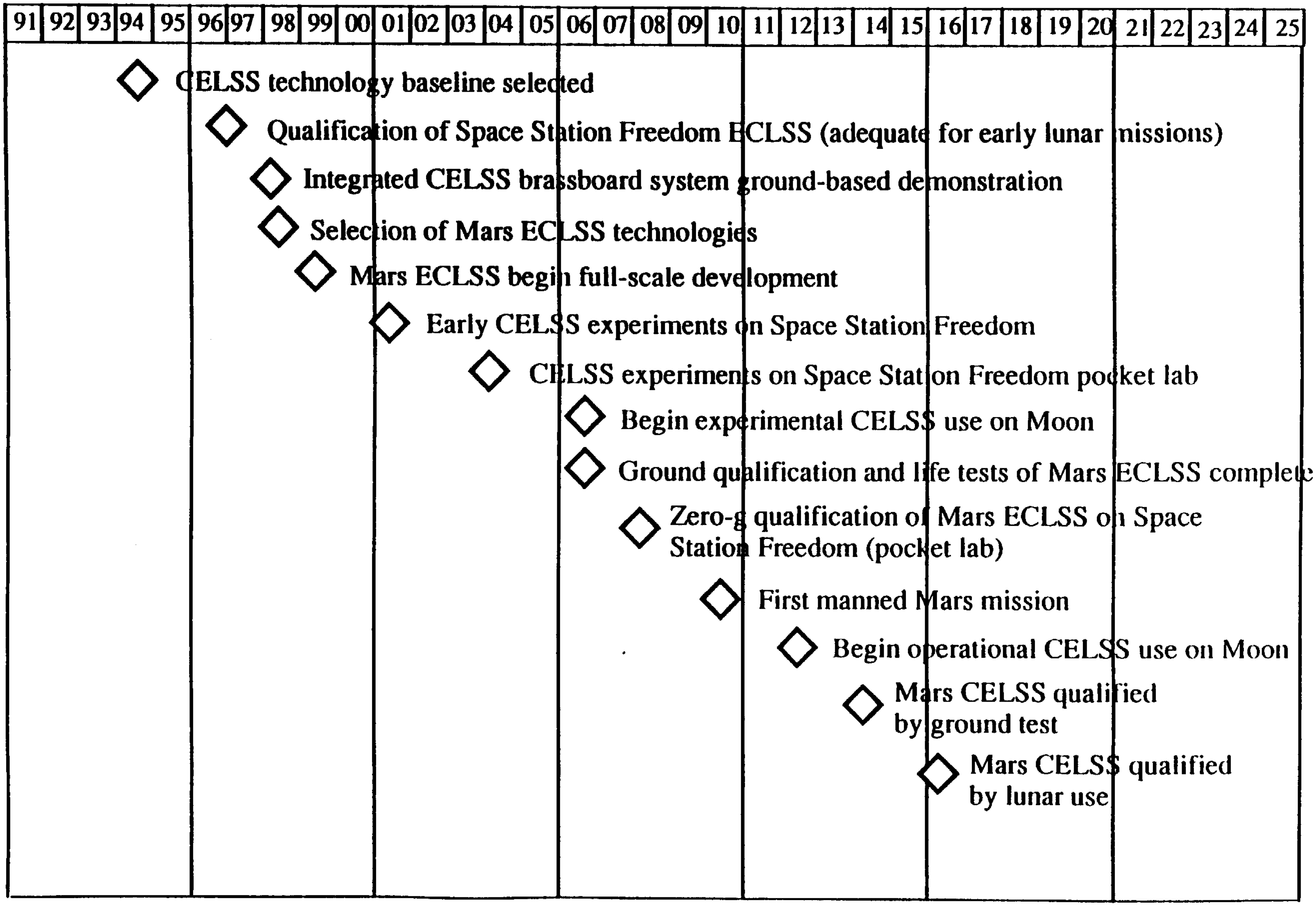


* If current technology is selected, no advancement activity is required.

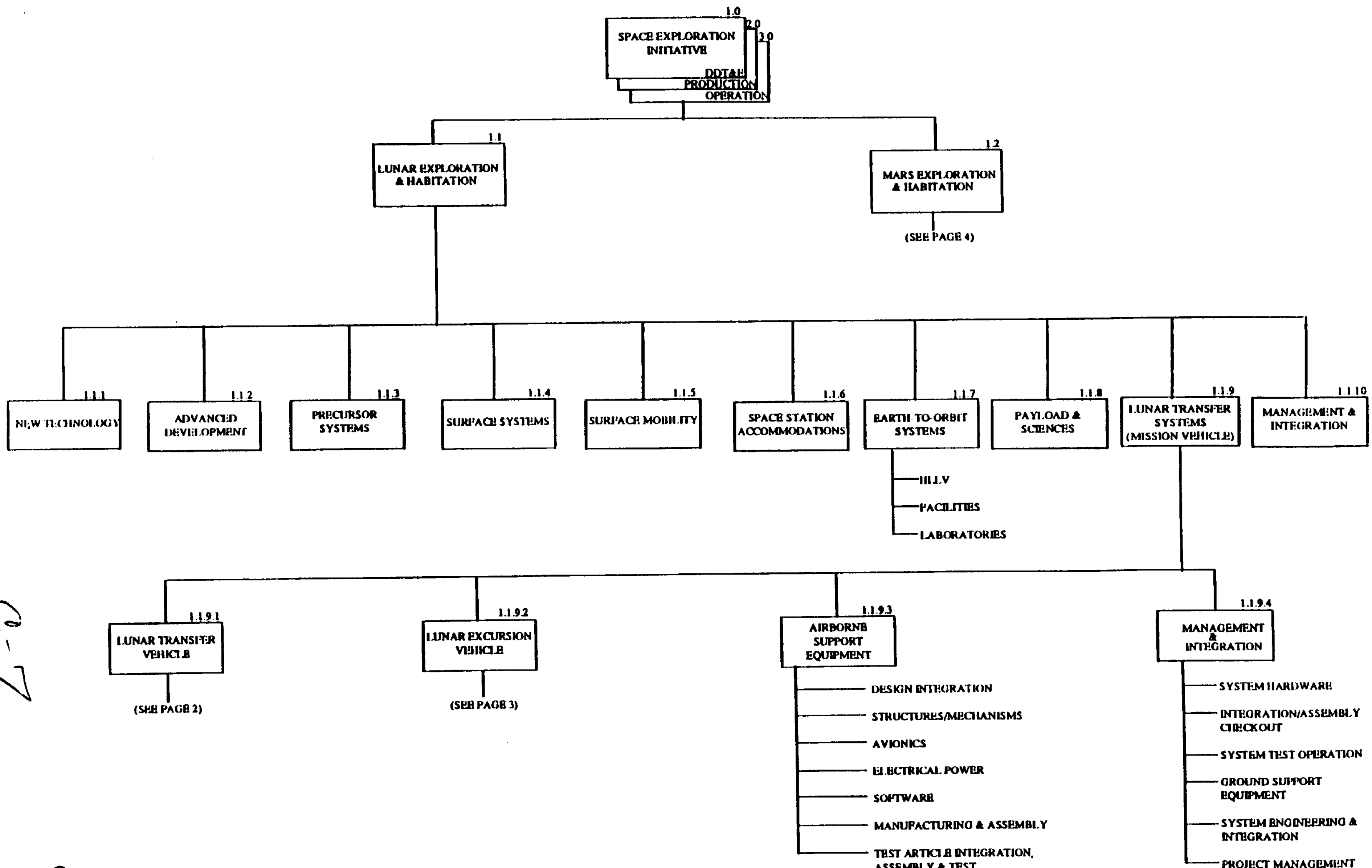
Avionics Major Test/Demo Man-Rating Approach



ECLSS Systems Man-Rating Approach



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



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SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

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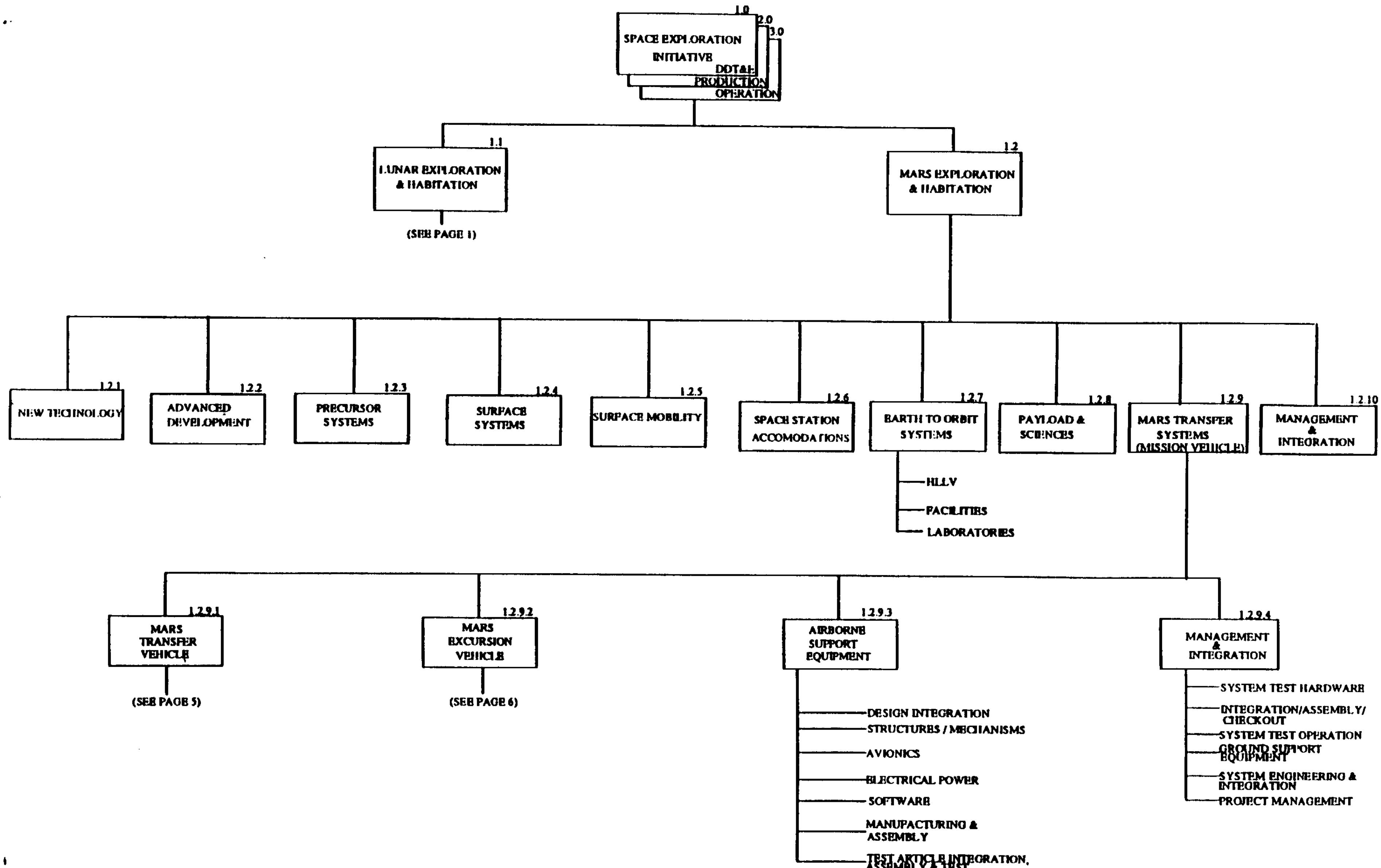
SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS



SPACE TRANSFER CONCEPTS AND ANALYSIS FOR EXPLORATION MISSIONS

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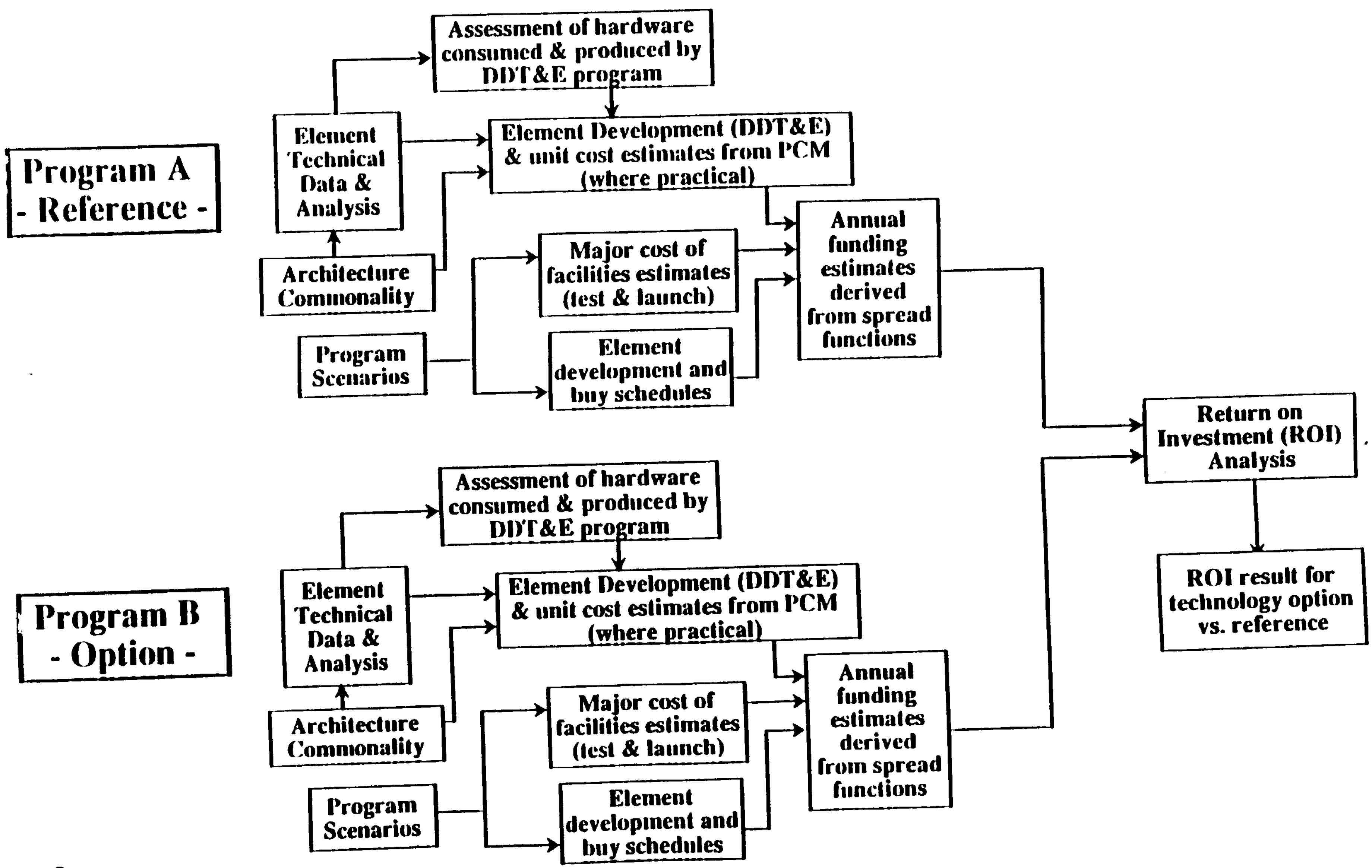
Costing Methodology Flow



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Boeing Parametric Cost Model (PCM)

Features

- Designed specifically for advanced system estimating
- Uses company-wide, uniform computerized data base
- Contains historical data compiled since 1969
- Allows direct input of known costs into the estimate

Main Inputs	Results
• Hardware Characteristics - Category (e.g., primary structure, power conditioning, etc.) - Weight (or Thrust) - Complexity - % Off-the-Shelf - Maturity - Quantity - Manufacturing Learning Curve • Support Cost Factors - Systems Engineering - Management - Operations - Spares	• DDT&E and Manufacturing Estimates - Based on previous Boeing programs - Provides first flight unit costs - Excludes test hardware - Excludes fees • New hardware must be relatable to PCM database to produce reasonable estimate • PCM estimates improve with increasing hardware detail.

LCCM Hardware Assignments

Components		Lunar/Mars		
		Minimum	Full Science	Settle /Ind
HLLV	Cargo Carrier & Core	X	X	X
	STME	X	X	X
	Recov PA Mod	X	X	X
Propulsion	Std Avionics Suite	X	X	X
	Adv Space Engine	X	X	
	NTR Tanks		X	
	MOC Tank	X		X
	MOC Core	X		X
	NTR Stage		X	
	NTR Engine		X	
	NEP Stage			X
	NEP Engine			X
	TMIS Engine	X		X
	TMIS Tank	X		X
	TMIS Core	X		X
Modules	LEO Tanker	X	X	X
	LTV Hab	X	X	X
	LTV	X	X	X
	LEV	X	X	X
	LEV Crew Module	X	X	X
	MTV	X		X
	MTV Crew Module	X	X	X
	MEV	X	X	X
	RMEV			X
	mini-MEV		X	
	MEV Crew Module	X	X	X
	Lunar Aerobrake	X		
	MTV Aerobrake			
	MEV Aeroshell	X	X	X
	MCRV	X	X	X

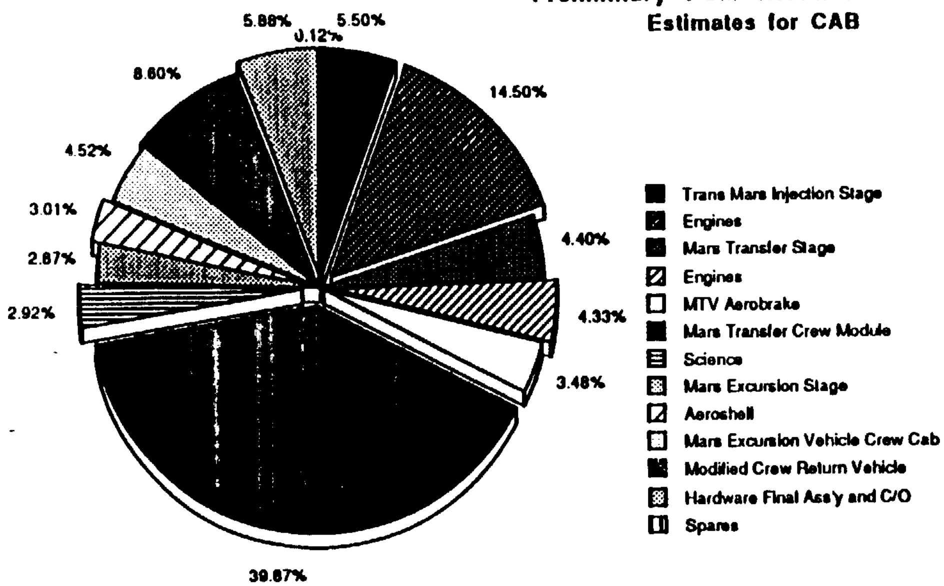
Mars CAB Preliminary PCM Summary

Item	Engineering ($Millions)	Manufacturing ($Millions)	Total ($Millions)
Trans Mars Injection Stage	86.379	220.784	307.163
Engines	710.078	99.868	809.946
Mars Transfer Stage	101.863	144.207	246.070
Engines	225.780	15.886	241.666
MTV Aerobrake	124.640	69.957	194.597
Mars Transfer Crew Module	1112.936	1114.365	2227.301
Science	100.651	62.517	163.167
Mars Excursion Stage	66.817	93.489	160.306
Aeroshell	112.107	56.312	168.418
Mars Excursion Vehicle Crew Cab	142.155	110.413	252.567
Modified Crew Return Vehicle	279.935	200.650	480.585
Hardware Final Ass'y and C/O	----------	328.266	328.266
Spares	----------	6.565	6.565
Hardware Total Costs	3063.339	2523.277	5586.613
System Engineering & Integration	516.245	----------	516.245
Software Engineering	361.606	----------	361.606
Systems Ground Test Conduct	2121.869	----------	2121.869
Systems Flight Test Conduct	----------	----------	----------
Peculiar Support Equipment	979.369	125.836	1105.204
Tooling & Special Test Equipment	----------	766.718	766.718
Task Direct Quality Assurance	----------	242.835	242.835
Logistics	156.132	----------	156.132
Liaison Engineering	268.181	----------	268.181
Data	64.806	----------	64.806
Training, Facilities Engineering, Safety, Graphics, Outplant, Program Management	O/H	----------	----------
Support Effort Total	4468.199	1135.389	5603.582
Total Estimate	**7531.539**	**3658.666**	**11190.203**

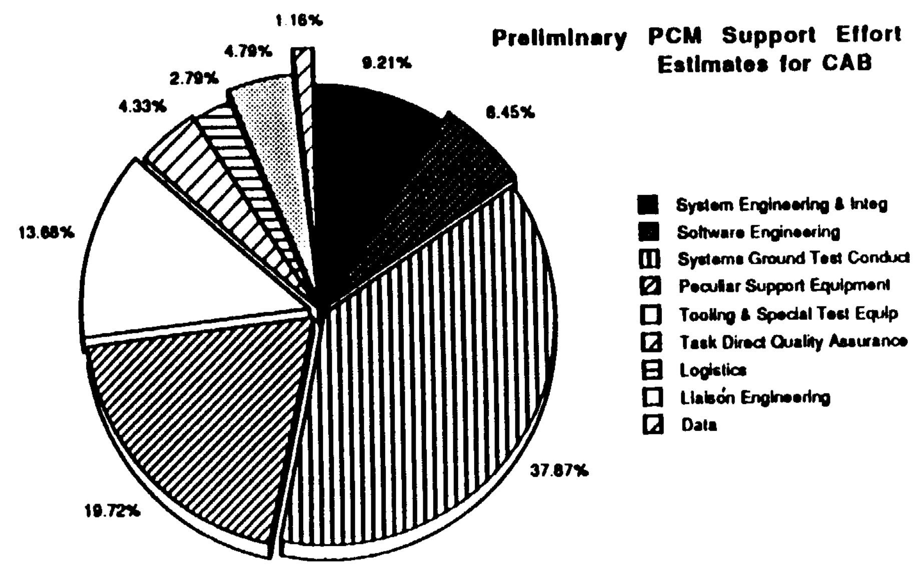
O/H = Overhead charge (included in above costs)

Mars CAB Preliminary PCM Summary - continued

Preliminary PCM Hardware Cost Estimates for CAB



Preliminary PCM Support Effort Cost Estimates for CAB



	A	B	C	D	E	F	G	H
1		Eng'r Cost	Cost Δ's	Wrap Factor	Total D&D	Unit Cost	# Units in DDT&E	
2	TMIS Core	86.4	0	2.708	233.9712	225	3.5	
3	TMIS Tank	8	0	2.708	21.664	42.5	3	
4	TMIS Eng	710	0	2.708	1922.68	26.8	12	
5	MTV	102	0	2.708	276.216	165.8	3.65	
6	MTV Eng	0	0	2.708	0	8	5	
7	Capture Brake	125	0	2.708	338.5	80.5	3	
8	Crew Module	1113	0	2.708	3014.304	1281	3.1	
9	MEV Stg	66.8		2.708	180.8944	107.5	3.5	
10	MEV Engine	0	0	2.708	0	8	5	
11	MEV Aeroshell	112.1		2.708	303.5668	64.7	2	
12	Mev CM	142		2.708	384.536	126.5	3.5	
13	BOCV	280		2.708	758.24	214	3.5	
14								

	I	J	K	L	M	N	O
1	DDT&E no Fe	Fee Factor, %	Total DDT&E	Units/Msn	Unit $/Msn	Msn Cost w/	Fee
2	1021.4712	8	1103.1889	1	225	243	
3	149.164	8	161.09712	4	170	183.6	
4	2244.28	8	2423.8224	5	134	144.72	
5	881.386	8	951.89688	1	165.8	179.064	
6	40	8	43.2	2	16	17.28	
7	580	8	626.4	1	80.5	86.94	
8	6985.104	8	7543.91232	1	1281	1383.48	
9	557.1444	8	601.715952	1	107.5	116.1	
10	40	8	43.2	7	56	60.48	
11	432.9668	8	467.604144	1	64.7	69.876	
12	827.286	8	893.46888	1	126.5	136.62	
13	1507.24	8	1627.8192	1	214	231.12	
14		Grand Total	16487.3258				2852.28

Development Risk Assessment For Aerobraking By Function

MISSION FUNCTION	BRAKE SIZE	ATMOSPHERE KNOWLEDGE & UNCERTAINTY	TARGET FOR ENTRY: GN&C PRECISION	HEATING/TPS	AERO PASS GN&C PRECISION REQUIRED
Lunar return Earth landing	Small, no ass'y required	Accurate knowledge, low uncert. effect	Very high	State-of-the-Art	State-of-the-Art
Lunar return Earth landing	Moderate requires assembly	Accurate knowledge, high uncert. effect	Very high	State-of-the-Art	Believed State-of-the-Art
Mars landing from orbit	Large, requires assembly	Poor knowledge, low uncert. effect	Can be high, e.g. done from Mars orbit	State-of-the-Art	Believed State-of-the-Art
Mars return Earth landing	Small, no ass'y required	Accurate knowledge, moderate uncertainty effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Accurate knowledge, high uncert. effect	Very high	Very high heating rates, TPS advancement needed	Believed State-of-the-Art
Mars return aerocapture	Large, requires assembly	Poor knowledge, high uncert. effect	Poor, unless nav-aids in Mars orbit	High heating rates, some TPS advancement needed	Advancements required

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ADVANCED CIVIL SPACE SYSTEMS

Cost Estimation Ground Rules

BOEING

- Parametric for Hardware Elements
- Each Major Elements = Developmental Project
- ETFU for Developmental Program
- Cost/Mass Identification/Parametricization
- No Contingencies (NASA to provide)
- Payload Costs Factored
- No Learning Curve for < 4 units/yr
- 15% Initial Spares +10% of Active Mass/yr of Service for Reusable or Long Life
- Mission Ops Support Factored Only
- Ground Ops Factored From Hardware Cost
- Lumped Cost Spread (eg DD T&E))
- SE&I & Management Costs Factored
- In Space Support Factored
- Limited To Through STV Integration (eg SE&I)
    - No Mission
    - No Carry on
    - No Launch Vehicles (will use estimates for necessary cost trades)
- PRICE - S for Software
- PRICE - H for Hardware
- Ground Support - $\$/FT^2$

Cost Analyses

Preliminary Work on:

- **MMV DDT & E**
- **MMV Manufacturing**
- **MMV Support (Manufacturing & DDT & E)**
- **ECCV Unit Manufacturing by Subsystem**
- **TMIS/MTV Manufacturing by Subsystem**
- **MEV Manufacturing by Subsystem**

First Release Scheduled for May 30, 1990